Institute of Space Systems
System Analysis Space Segment

Feasibility Study

Post-ISS Scenario-I

Concurrent Engineering Study Report

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Study Team and Responsibilities

Pos.	Domain	Responsible	Institution
	Costumer/ Project Management	S. Baerwalde/ M. Waßmann	FS-R (DLR-KP)/ PD-R (DLR-KP)
	Project Leader system studies Post-ISS	O. Romberg	RY-SRS (DLR-HB)
	Team Leader	D. Quantius / V. Maiwald	RY-SRS (DLR-HB)
	Operations / Crew Support / Regulatory	M. Gold	Bigelow Aerospace
	Data Model support	P. Fischer	SC-SRV (DLR-BS)
1	Systems / Cost	A. Braukhane / D. Quantius	RY-SRS (DLR-HB)
2	Thermal	K. Sasaki	RY-SEK (DLR-HB)
3	Power	E. Wejmo / J.-T. Grundmann / G. Zamka	RY-SRS (DLR-HB) / RY-SPB (DLR-HB) / Bigelow Aerospace
4	Crew Facilities / ECLSS	H. Schlegel	ESA (NASA)
5	AOCS	S. Theil / B. Lovrinovic	RY-GNC (DLR-HB)/ RWTH
6	Propulsion / Launch Scenario	V. Clark	RY-SRT (DLR-HB)
7	Configuration	V. Vrakking	RY-SRS (DLR-HB)
8	Structure / Debris & Rad. Protection	J. Romero (W. Bauer)	RY-SRS (DLR-HB)
9	Robotic / Automation / Mechanisms	S. Jäkel	RM-AUF (DLR-OP)
10	Mission Analysis	M. Hallmann	RY-SRS (DLR-HB)
11	Comm. / Ground Segment / OBC	A. Martelo	RY-SRS (DLR-HB)
12	Payloads / Science	W. Seboldt	RY-SRS (DLR-HB)

DLR Deutsches Zentrum für Luft- und Raumfahrt
German Aerospace Center

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Deutsches Zentrum für Luft und Raumfahrt e.V.
in der Helmholtz-Gemeinschaft

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Edited by:
Dominik Quantius

Robert-Hooke-Str. 7
D-28359 Bremen
Telefon 0421 24420-1109
Telefax 0421 24420-1120
E-Mail mailto: dominik.quantius@dlr.de
Internet http://www.dlr.de/irs/en/

Table of Contents

List of Figures

List of Tables

1. Introduction

All Space Station partners agree to utilise the orbital research facility until 2020. NASA, Roscosmos and CSA announced to utilise the ISS even further until 2024. Whether this is politically, technologically and financially feasible for all partners is unknown. The overall question for all users is whether and how to continue with their investments. In general a transition to a new concept without critical know-how losses is around 10 to 15 years. Therefore, from a German/European point of view the technical layout, road mapping and development of a Human Spaceflight concept must be started now. The DLR project "Post-ISS" (system analysis study) can be understood as national preparatory work for establishing future programmes in the field of Human Spaceflight securing long-term research and astronautical activities in LEO. Corresponding questions focus on:

- How to continue with space research and space technology development after the ISS utilisation period (≥~2024)?

Therefore, the following objectives have been defined within the DLR study:

- Analysis of the ISS pros and cons (DLR internally) and recommendations based on Lessons-Learnt
- Market research of existing technologies / techniques
- Analysis of additional user demand and utilisation opportunities by including additional scientific disciplines and technological research
- Design of user conform infrastructure concepts to proceed with Human Spaceflight in LEO
- Analysis of re-usability of current architecture

The Concurrent Engineering (CE) study "Post-ISS Scenario-I" took place from 8th to 12th of June 2015 in the Concurrent Engineering Facility (CEF) at the DLR Bremen. The subsystem domains and disciplines were taken by Bigelow Aerospace, Consultants and mainly DLR Bremen staff. The goal of the study has been the investigation of the Base Station concept developed in the frame of the DLR-internal Post-ISS project.

1.1. General Background

Since decades the International Space Station ISS demonstrates not only long-term international cooperation between 14 partner governments but also a significant engineering and programmatic achievement mostly as a compromise of budget, politics, administration and technological feasibility. Most ISS technologies are based on

MIR and other previous experience. Due to high safety standards required for human space activities, latter are often conservative and new developments require patience and waiving 'state-of-the-art' technologies. A paradigm shift to more innovation and risk acceptance can be observed in the development of new markets by shifting responsibilities to private entities and broadening research disciplines, demanding faster access by users and including new launcher[1] and experiment facilitator companies[2] (see U.S.).

The research part of the systems-engineering study shows that space faring nations are developing their individual programmes for the time after ISS: NASA shifts LEO operations and utilisation to competing U.S. commercial companies while focussing on the next preparatory steps of Exploration (e.g. SLS, MPCV) of Asteroids, Moon and in long-term Mars. Russia plans new human rated space infrastructures at various optional locations (e.g. OKA-T Free-Flyer) rather than committing to continue the utilisation of its dated ISS modules. In the field of human spaceflight China proceeds to go on with its Chinese Space Station CSS and prepares its next objective: the human Moon landing. Europe's human spaceflight partners seem to tend to the consideration of new platforms in LEO or cis-lunar space while utilising ISS as long as possible and necessary for the transition expected beyond 2024. Europe itself is interested in LEO and Human Spaceflight as discussed in ISECG, depending on the funding commitment. [RD-1]

In line with the space strategy of the German Government ISS follow-on activities should comprise of clear scientific objectives and technological key competences (e.g. robotic, internal and external structures, module/facility and experiment operations, interface systems (ATV)).

Therefore, DLR started to investigate future options by evaluating various LEO infrastructure concepts including opportunities for national realisation or international cooperation. A corresponding list of options can be found below. DLR scientists from various disciplines were asked to assess the usability of these options and design payloads based on their MIR and ISS experience and with respect to future scientific fundamental and technological research questions.

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[1] US commercial launch providers currently are for example: SpaceX, Orbital Sciences.

[2] European experiment facilitators AIRBUS and OHB tried the commercial approach but are still waiting for the success. US experiment facilitators are for example: Nanoracks, Kentucky Space and the mediator foundation CASIS. The only platform provider with a commercial approach is Bigelow.

1.2. Mission Outline

1.2.1. Mission Objectives

In the frame of the Post-ISS project, initiated by the DLR executive board, the following question shall be answered: *Assumed, Germany or Europe wants to continue the astronautic spaceflight in LEO: How could options look like, whilst considering the scientist's requirements?*

1.2.2. Mission Goals

The requirements of the science community have been defined during the Post-ISS Payload CE-Study based on the User-Workshop (Cologne, May 2014). Now, these requirements shall be addressed in a more detailed architecture design. For that purpose the following shall be elaborated during the Post-ISS Scenario-I CE-study:

- Distribution of needed functions over modules (e.g. communication, ECLSS)
- Positioning / orientation of modules
- Layout of modules (primary structure and secondary structure, harness, accommodation, power, subsystems (including scientific payloads))
- Integration of robotic / automation
- Operations scenario
- Design of infrastructure on ground
- Installation scenario / launch
- Rough cost estimation

Thereby the following framework conditions shall be considered:

- Technical modular concept (separation of astronauts and experiments where required by science restrains; in failure case single modules' exchange is possible, optional autonomous operation of units (Habitat/ temporarily crewed Free-Flyer)
- Political modular concept (countries/agencies can participate according to individual budget possibilities and science interest)
- Design (mainly) based on available technologies with participation of private partnerships
- User (science) requests for multiple disciplines (see details below)
- Reasonable costs for operations

1.3. Concurrent Engineering Approach

To investigate and define the technical concept of the Post-ISS Scenario-I a Concurrent Engineering (CE) Study at DLR Bremen has been conducted. The CE-study comprised the analysis and the development of all subsystems necessary for Post-ISS Scenario-I i.e. Systems, Cost, Thermal, Power, Crew Facilities, ECLSS, EVA, AOCS, Propulsion, Launch Scenario, Configuration, Structure, Robotic, Mission Analysis, Communication, OBC and Science.

The applied Concurrent Engineering (CE) process is based on the optimization of the conventional established design process characterized by centralized and sequential engineering (see Figure 1-1 top). Simultaneous presence of all relevant discipline's specialist within one location and the utilization of a common data handling tool enable efficient communication among the set of integrated subsystems (see Figure 1-1 bottom).

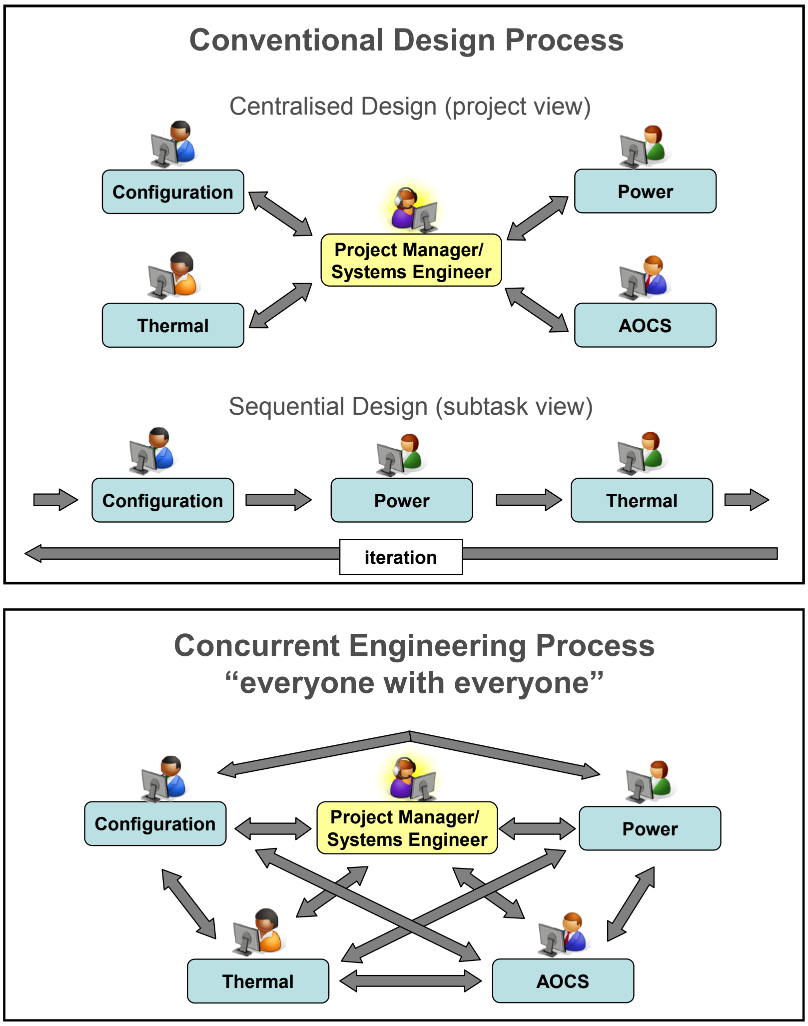


Figure 1-1: The Concurrent Design approach compared to projections of conventional design process.

The CE-Process is based on simultaneous design and has four phases ("IPSP-Approach"):

1. Initiation Phase (starts weeks/months before using the CE-facility):

- Customer (internal group, scientists, industry) contacts CE-team
- CE-team-customer negotiations: expected results definition, needed disciplines

2. Preparation Phase (starts weeks before using CE-facility):

- Definition of mission objectives (with customer)
- Definition of mission and system requirements (with customer)
- Identification and selection of options (max. 3)
- Initial mission analysis (if applicable, e. g. based on STK)
- Final definition and invitation of expert ensemble, agenda definition

3. Study Phase (1- 3 weeks at CE-Facility in site):

- K/O with presentations of study key elements (goals, requirements)
- Starting with first configuration approach and budgets estimates (mass, power, volume, modes, …) on subsystem level
- Iterations on subsystem and equipment level in several sessions (2- 4 hours each); trading of several options
- In between offline work: subsystem design in splinter groups
- Final Presentation of all disciplines / subsystems

4. Post Processing Phase:

- Collecting of Results (each S/S provides Input to book captain)
- Evaluation and documentation of results; transfer open issues to further project work

The DLR's Concurrent Engineering Facility in Bremen is derived from the Concurrent Design Facility at ESA's ESTEC (European Space Research and Technology Centre), which has already been in operation for more than ten years. The CEF has one main working room where the whole design team can be assembled and each discipline is supplied with an own working with special design tools and a common design and data model. Three screens allow display of data in front of the team. Further working positions are provided in the center of the working area and are usually reserved for customers, advisors, guests as well as the team leader. Two more splinter rooms provide the design team with separated working spaces where sub-groups can meet, discuss and interact in a more concentrated way.

Figure 1-2: Concurrent Engineering Facility main room during CE-study phase at DLR Bremen

The major advantages of the Concurrent Engineering (CE)-process are:

- Very efficiency regarding time, cost & results of a design activity
- Assembly of the whole design team in one room facilitates direct communication and short data transfer times, supported by a moderator
- The team members can easily track the design progress, which also increases the project identification
- Ideas and issues can be discussed in groups, which brings in new viewpoints and solutions; incl. avoidance and identification of failures and mistakes

1.4. Document Information

This document summarizes the progress and results of the DLR Concurrent Engineering study about the Post-ISS Scenario-I, which took place from 8th to 12th of June 2015 in the Concurrent Engineering Facility of the DLR Institute of Space Systems in Bremen. The single subsystems or domains as investigated during the study are covered in individual chapters, which explain the study progress, elaborate on decisions and trade-offs made during the study and also design optimizations. If not allowed by the DLR directorate, the document should not be distributed outside DLR before June 2017 (study team members excluded). A comprehensive documentation of the overall Post-ISS project in German can be found under *DLR-RY-Post-ISS Projektbericht: AP1000 "ISS-Analyse und Lessons Learnt"*; *DLR-RY-Post-ISS Projektbericht: AP2000 "Konzept-bewertung"*, *DLR-RY-Post-ISS Projektbericht: AP3000 "Mögliche Anwendungen & Nutzlasten"*, *DLR-RY-Post-ISS Projektbericht: AP4000 "Szenarienentwurf"*.

2. Systems

2.1. Mission Requirements

In preparation of the CE-study the following mission requirements have been defined:

Table 2-1: Post-ISS Scenario-I – Mission Requirements.

No.	***Requirements***
MI-010	The station shall maintain an orbit altitude of 400 km +/- 50 km
MI-020	The station shall maintain an orbit inclination of about 51.6 deg
~~MI-030~~	~~The Base Station shall be oriented nadir~~

Hereby the 51.6° inclination was considered "bad" in terms of beta-angle and launch (from a KSC/ Kourou point of view) but welcomed by the observation science, Baikonur capability. Also it fits best to the well-established and proven ground network. As a consequence, diverting from 51.6° would need major changes in the communication architecture.

The nadir requirement for the Base Station (MI-030) was dropped during the study, because all pointing-critical items (e.g. for Earth observation) are foreseen for the Free-Flyer only. Without the nadir constraint, the Base Station can be turned or rolled in order to point the solar panels towards the Sun for beta angle corrections (see section 8.2). This allows for a much simpler design of the solar panels (i.e. only one axis rotatable) in comparison to the solar panels of the ISS mounted on the truss structure.

2.2. System Requirements

In order to dimension the spacecraft's subsystems the following system requirements have been defined and fulfilled by the design team:

Table 2-2: Post-ISS Scenario-I - System Requirements.

No.	***Requirements***
ST-010	The design shall be based on technologies that are available 2025
ST-020	Each module shall fit to today available launchers (mass, size)
ST-030	The international docking standard shall be used (IBDM: ∅80 cm)
ST-040	As part of Option A.4 the Base Station shall be able to dock with Free-Flyer and provide ECLSS for the pressurized part of the Free-Flyer
ST-050	The station shall be laid out for a crew of up to three persons continuously in Base Station (temporarily more depending on transport vehicle) and temporarily up to two persons in pressurized part of docked Free-Flyer

2.2.1. Science Requirements

For the following subjects in section 4 the science needs are described:

- Human physiology (measurement of intracranial pressure)
- Radiation dosimetry and biology (e.g. Phantom)
- Gravitation biology (signal transduction)
- Robotic Experiments
- Shared Equipment
- Technology Demos
- Future Science placeholder

2.3. Baseline Design

The laid out concept strives to employ only the minimum functionality required for a scientific astronautical base station (three crew members continuously plus visitors) in LEO: At least one module is needed for science laboratories, the crew accommodation and according environmental control and life support systems (*Habitat; HA*). In addition, a service module is needed to ensure attitude and orbit control and provide power and thermal control (*Service Module; SM*). A five-point docking node (one used by the cupola) allows for crew and cargo transfer and extension opportunities and can comprise communication and data systems or backup subsystems (*Docking Node; DN*). Up until today, there have been 187 EVAs on the ISS. In contrast to the ISS, the Post-ISS Scenario-I concept is designed to limit the number of EVAs by avoiding items placed externally to the station. However an EVA contingency is foreseen on the Base Station and an airlock is planned for the pressurized part of the Free-Flyer in order to service the External Science Platform using a robotic arm. Since the critical requirements regarding attitude and disturbances are shifted towards the Free-Flyer, the Base Station is free to roll or yaw a certain amount. That allows for a one-axis rotatable solar panel design which does not need additional truss structures as used on the ISS. The Base Station is also free to have the Habitat module or the Docking Node point into the direction of flight. To avoid regular refuelling for orbit maintenance, the respectively docked crew or cargo vehicle will provide the required manoeuvres. Hereby electrical thrusters are a promising solution for drag compensation.

Table 2-3: Subsystem distribution[3] per Module.

Subsystem	Habitat	Service Module	Docking Node
On-Board Computer	x	x	
Communication		x	x
Crew Facilities	x	x	x
ECLSS[4]	x	x	x
EVA	x		
Science Base Station	x		
AOCS	x	x	x
Power[4]	x	x	
Thermal[4]	x	x	x
Propulsion[3]	x		x
Structure	x	x	x

[3] From subsystem point of view the respective units are allocated to the different modules. Propellant is listed separately.
[4] ECLSS (subsystems), Harness (Unit), Ammonia Coolant pipe (Unit) equally distributed over three modules.

2.3.1. Power Modes

During the study the following operational modes have been defined in order to lay out the power subsystem. Since chemical propulsion for drag compensation against electrical propulsion was not predetermined, there are two standard modes which differ in the propulsion concept.

Table 2-4: Modes of Operation for Post-ISS Scenario-I.

Mode Name	Abbreviation	Description	Reference Duration
Standard Mode (Default)	StM	Core operations, crew onboard, no reboost activities; pitch down for balancing drag; separated from free-flyer	2 weeks
Standard Mode, electrical	StMe	Core operations, crew onboard, continuous reboost by electrical propulsion; pitch down for balancing drag; separated from free-flyer	2 weeks
Crewexchange Mode	CrM	Core operations, two crews onboard for handover (i.e. two vehicles attached)	1 week
Docked Mode	DoM	Core operations, crew onboard, free-flyer attached and ECLSS connected	1 week
Survival Mode	SuM	(Loss of attitude control, resolving of fire situation, leakage, loss of power) ECLSS running, crew onboard, reboost capability is retained, power consumption minimized	4 days
Proximity Operations Mode	PoM	Nadir pointing, horizontal attitude, solar array moved into a safe position (minimize collision risk), chemical propulsion	1 day

2.3.2. Overall Power budget

In the following table the power budget for all domains and power modes is given. Finally in the "Standard Mode electrical" the additional power need for the electrical engine was not regarded. However the main solar panels are laid out for the numbers given in Table 2-5 and additional power can be provided by solar panels which will be attached to the Habitat.

Table 2-5: Power budget of Post-ISS Scenario-I.

System Component	Parameter	Unit	Default	Standard_elect...	CrewExchange	Docked	Survival	Proximity_Ops
BaseStation	mode_duration	Second	1209600.000	1209600.000	604800.000	604800.000	345600.000	86400.000
BaseStation	power_avg_wMargin	Watt	25090.875	25090.875	27217.811	25090.875	12430.108	25103.195
AOCS	power_avg_wMargin	Watt	286.495	286.495	286.495	286.495	255.516	298.815
Communication	power_avg_wMargin	Watt	781.200	781.200	781.200	781.200	781.200	781.200
CrewFacilities	power_avg_wMargin	Watt	320.240	320.240	320.240	320.240	179.440	320.240
OnBoardComputer	power_avg_wMargin	Watt	4442.400	4442.400	4442.400	4442.400	2810.400	4442.400
Power	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
Propulsion	power_avg_wMargin	Watt	1364.780	1364.780	1364.780	1364.780	1364.780	1364.780
Robotic_Mechanisms	power_avg_wMargin	Watt	27.000	27.000	27.000	27.000	0.000	27.000
Science_on_BaseStation	power_avg_wMargin	Watt	7974.576	7974.576	7974.576	7974.576	761.520	7974.576
Structure	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
Thermal	power_avg_wMargin	Watt	2801.280	2801.280	2801.280	2801.280	2801.280	2801.280
ECLSS	power_avg_wMargin	Watt	7092.904	7092.904	9219.840	7092.904	3475.972	7092.904
EVA	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
VisitingVehicles	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
BaseStation	power_avg_wMargin_wS...	Watt	30109.050	30109.050	32661.373	30109.050	14916.129	30123.834

2.3.3. Overall Mass budget

Within at least six domains the margins for the equipment maturity was set to the maximum value of 20%. That does probably more reflect uncertainties caused by the early design stage than based on the components' technology readiness level. Furthermore the systems margin of 20%, which is common for satellite projects, results in more than 10 t of mass in addition which might decrease a bit during later design phases.

Table 2-6: Mass budget of Post-ISS Scenario-I.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
AOCS	1166.60	10.21	119.09	1285.69	2.43
Communication	101.00	20.00	20.20	121.20	0.23
CrewFacilities	1310.00	10.48	137.30	1447.30	2.74
ECLSS	3150.00	17.33	546.00	3696.00	6.99
EVA	730.00	12.05	88.00	818.00	1.55
OnBoardComputer	984.00	20.00	196.80	1180.80	2.23
Power	3593.80	20.00	718.76	4312.56	8.16
Propulsion	742.90	12.40	92.11	835.01	1.58
Robotic_Mechanisms	127.00	20.00	25.40	152.40	0.29
Science_on_BaseStation	2750.80	20.00	550.16	3300.96	6.24
Structure	25312.92	18.20	4607.58	29920.50	56.60
Thermal	4830.00	20.00	966.00	5796.00	10.96

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	44799.02			52866.42	
System margin:		20.00		10573.28	
Total dry mass with system margin:				**63439.71**	
Propellant:				1752.40	
Adapter mass:				125.00	
Launch Mass:				**65317.11**	

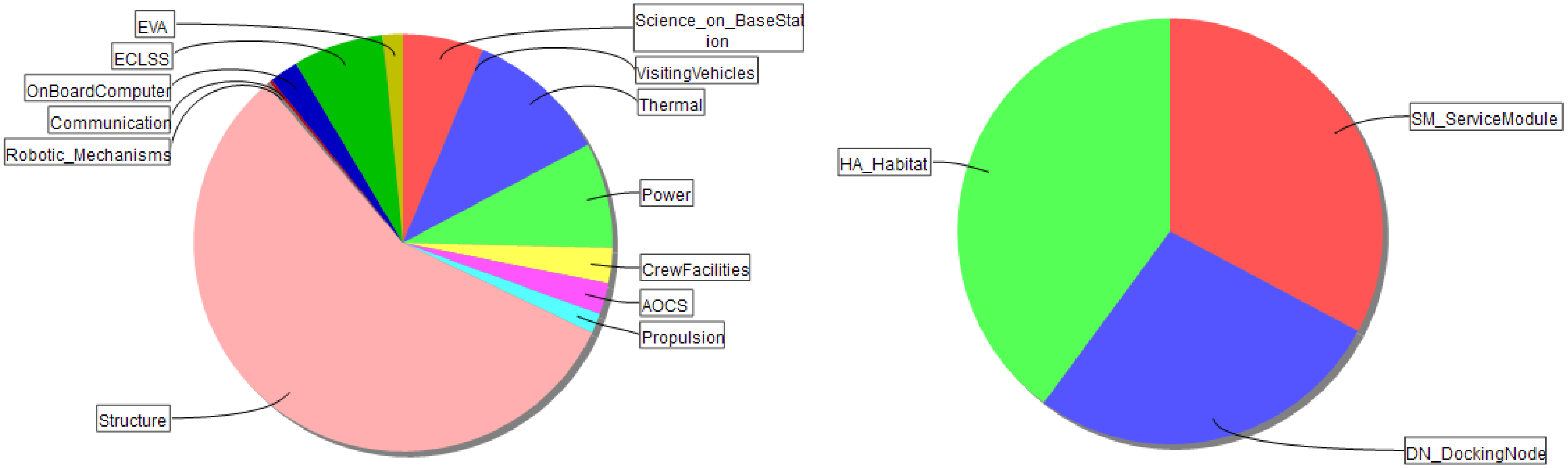


Figure 2-1: Mass distribution of Post-ISS Scenario-I (left: overall mass; right: mass per module).

Table 2-7: Mass budget per module for Post-ISS Scenario-I.

Module	Mass [t]	w/ 20% [5]	Propellant	Launch[6]
HA	21,0	25,2	0,81	26,1
SM	17,3	20,7	0,94	21,8
DN	14,4	17,3	-	17,4
Total	52,7			65,3

[5] 20% system margin (TBC)
[6] incl. 125 kg launch adapter (here w/c assumption → 3 launches)

2.4. To be studied / additional Consideration

- Margin selection on equipment level
- Margin approach on systems level
- Combination/functional distribution between Service Module and Docking Node
- Preferred flight direction

2.5. Summary

During the study week a basic orbital station for a continuous crew of three was laid out. Due to the fact, that all items that require a fixed pointing, like e.g. observation instruments, will be installed on a Free-Flyer, the Base Station was more relaxed in terms of attitude constraints. Utilising that freedom, a just one axis rotatable solar panel design made the concept leaner compared to the ISS. Another approach which reduces the cost of the station is that it is not designed for EVAs. Only a contingency air lock is remained for the astronauts and a payload air lock is planned for the Free-Flyer. That means that the station's assembly has to be automated and no parts will be placed outside the modules. The overall concept would not need more than three launches for the Base Station when the Free-Flyer already was in orbit. It is a minimum astronautical station with a high degree of modularity and extension opportunities. It can serve as a hub, where spacecraft (like the Free-Flyer) can dock and be serviced, or goods (e.g. propellant or experiments) can be distributed (cf. hub as distribution node of the Internet).

3. Configuration

3.1. Requirements and Design Drivers

The following requirements were given influencing the configuration:

- Each module shall fit to today available launchers (mass, size)
- The international docking standard shall be used (IBDM: Ø 80 cm)
- The station shall be laid out for a crew of up to three persons continuously in Base Station (temporarily more depending on transport vehicle) and temporarily up to two persons in pressurized part of docked Free-Flyer
- Base Station shall be able to dock with Free-Flyer

Furthermore there was the following framework condition defined:

- Technical modular concept (separation of astronauts and experiments where required by science restrains; in failure case single modules' exchange is possible, optional autonomous operation of units (Habitat/ temporarily crewed Free-Flyer)
- Political modular concept (countries/agencies can participate according to individual budget possibilities and science interest)
- Design (mainly) based on available technologies with participation of private partnerships
- User (science) requests for multiple disciplines
- Reasonable costs for operations

3.2. Baseline Design

3.2.1. Over-all System Configuration

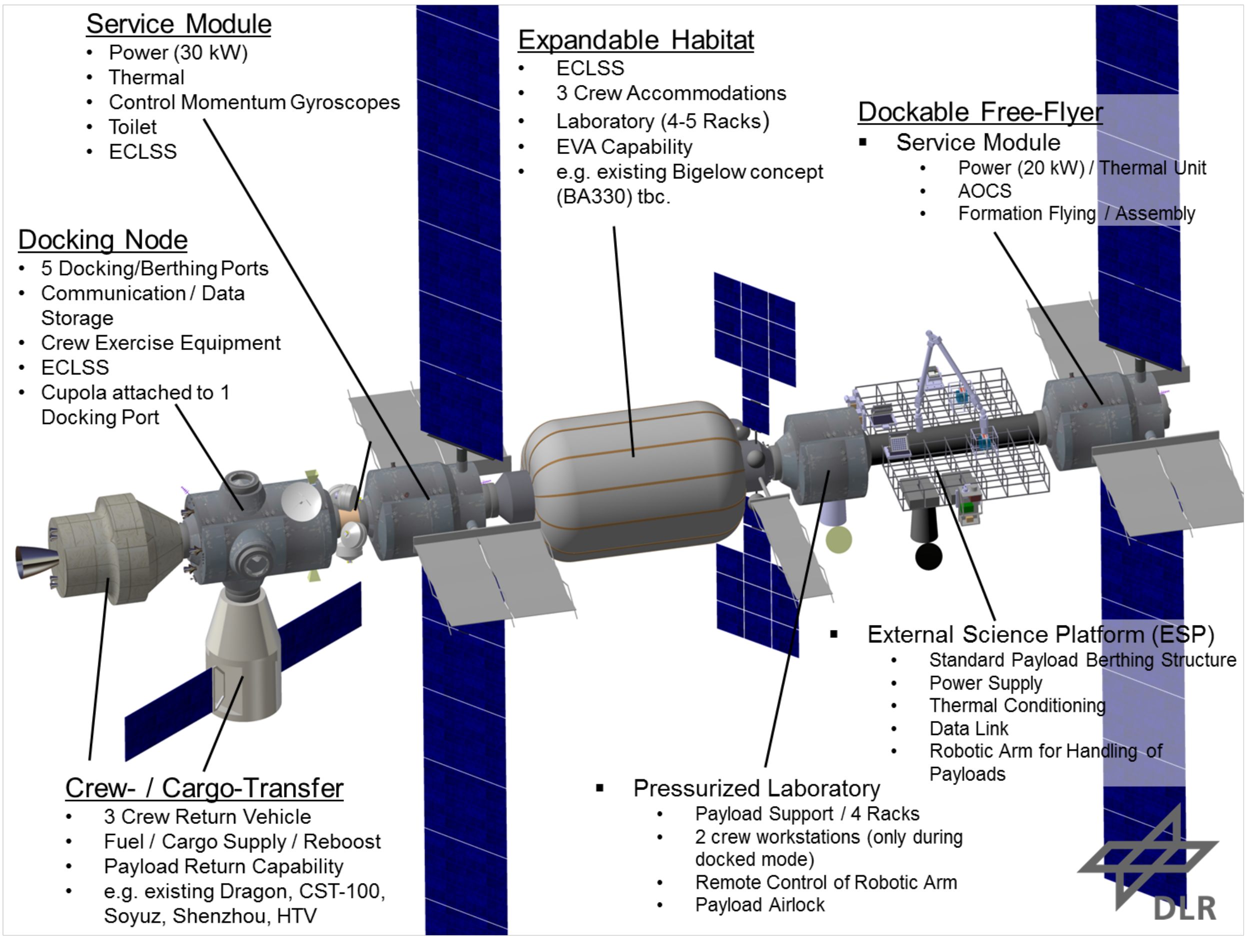


Figure 3-1: Configuration of Post-ISS Scenario-I.

3.2.2. Base Station

The selected concept strives to employ only the minimum functionality required for a scientific astronautical base station (three crew members continuously plus visitors) in LEO: At least one module is needed for science laboratories, the crew accommodation and according environmental control and life support systems (example design: expandable habitat). In addition, a service module is needed to ensure attitude and orbit control and provide power and thermal control. A five-point docking node (one used by the cupola) allows for crew and cargo transfer and extension opportunities and can comprise communication and data systems or backup subsystems. Up until today, there have been 187 EVAs on the ISS. In contrast to the ISS, the Orbital-Hub concept is designed to limit the number of EVAs by avoiding items placed externally to the station. However an EVA contingency is foreseen on the Base Station and an airlock is planned for the pressurized part of the Free-Flyer in order to service the External Science Platform using a robotic arm.

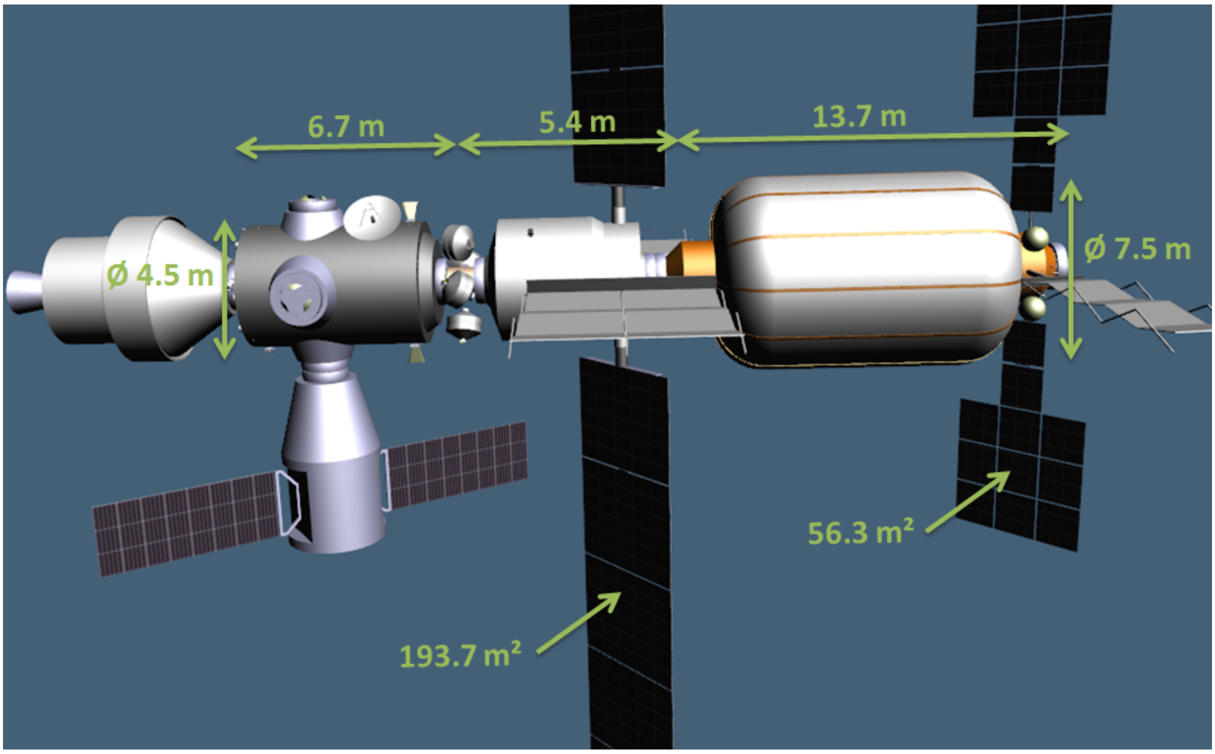


Figure 3-2: Post-ISS Scenario-I Base Station dimensions.

Since the critical requirements regarding attitude and disturbances are shifted towards the Free-Flyer, the Base Station is free to roll or yaw a certain amount. That allows for a one-axis rotatable solar panel design which does not need additional truss structures as used on the ISS. The Base Station is also free to have the Habitat Module or the Docking Node point into the direction of flight. To avoid regular refuelling for orbit maintenance, the respectively docked crew or cargo vehicle will provide the required manoeuvres. Hereby electrical thrusters are a promising solution for drag compensation.

3.2.3. Free-Flyer

In addition to the Base Station, a Dockable Free-Flyer is part of the Orbital-Hub concept in response to the scientific user requirements. It is intended to fly uncrewed in a safe formation to the Base Station for e.g. three-month periods until it can be maintained or reconfigured when docked to the station for short duration. Therefore in analogy to the Base Station, it also needs a service module for attitude and orbit control and also for formation flying and independent power and thermal control. It further contains a pressurised module for µg-research which can be accessed when docked to the Base Station (e.g. via the Docking Node or via the Expandable Habitat module) or to a crew vehicle. The external science platform is the centre of the Free-Flyer. It has a berthing structure for any external payload and provides power, data and thermal conditioning. The Free-Flyer will most likely fly with the instruments pointed nadir, but in principle, is free to change attitude for certain periods depending on user requirements. The size and shape of this platform is only an example and it is intended to be deployable in order to launch the Free-Flyer in one piece. Robotic arm interfaces are foreseen to handle the payloads on the platform, which is based on the Orbital-Hub User Concurrent Engineering study, described above. The Free-Flyer is intended to support the assembly of the Base Station by being the active part of automated docking, since there is currently no similar vehicle like the U.S. Space Shuttle available.

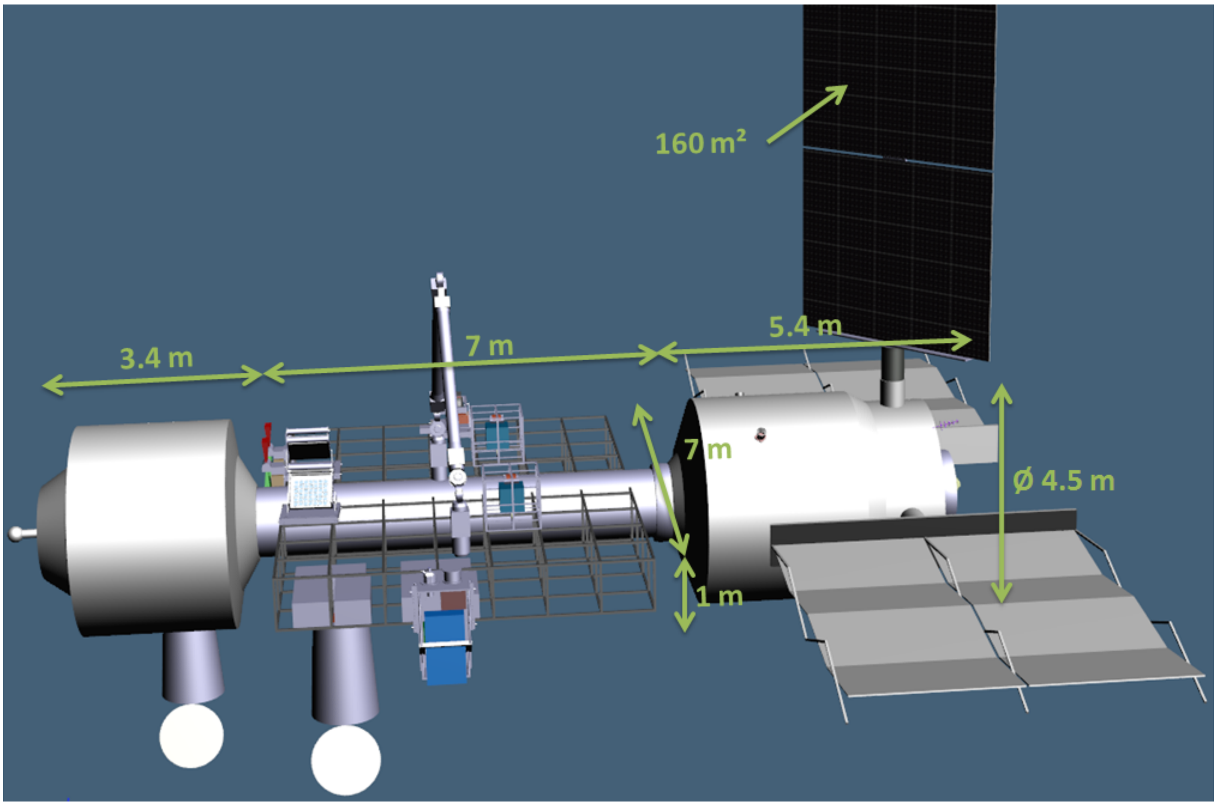


Figure 3-3: Post-ISS Scenario-I Free-Flyer dimensions.

3.2.4. The Bigelow Aerospace Module

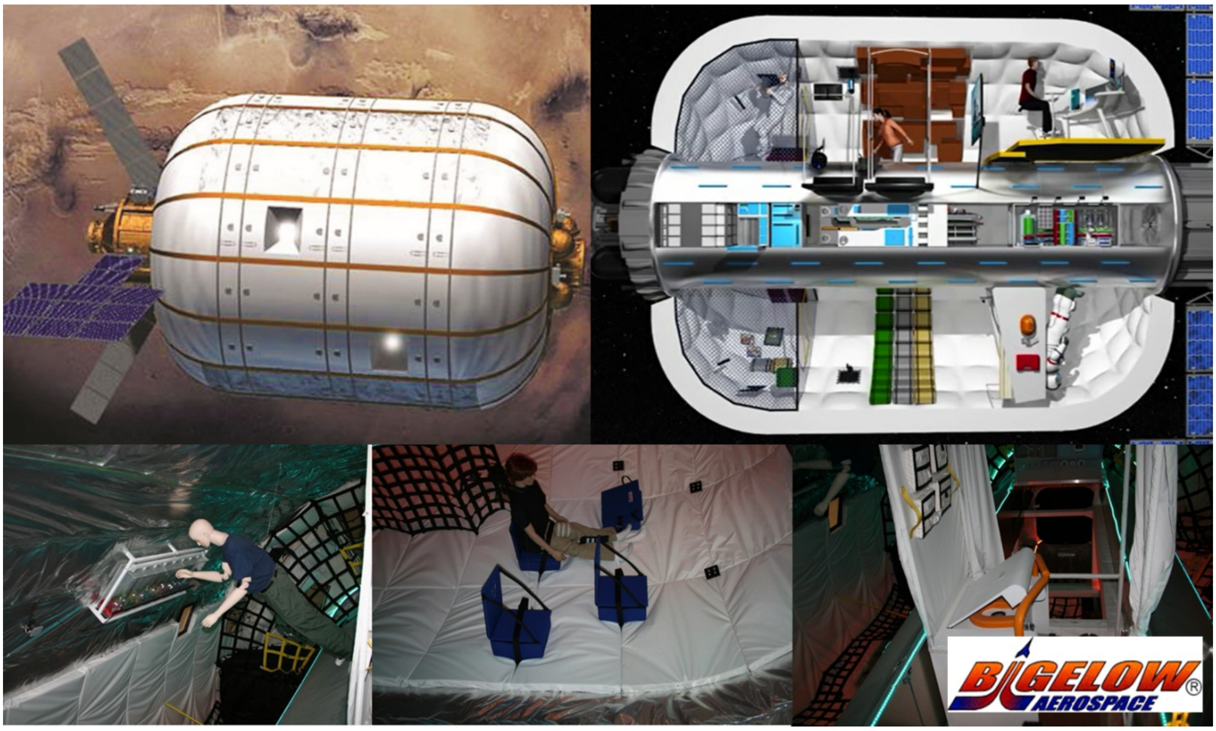


Figure 3-4: Bigelow Aerospace B330 concept.

- The expandable B330 habitat connects to other modules via the NDS system at both ends.
- The soft goods are attached to the internal truss system.
- The truss maintains the longitudinal dimension of B330 during launch
- On-orbit the expanded soft-goods hull is rigid.

Initial customer has the opportunity to identify launch upmass space, volume and support requirements/desirements

- Within chases
    - Access to vacuum venting, fluid lines, vibration isolation
- Mid-bays
    - Access to structural support from chase assemblies
- Spacecraft core
    - Structural support options easiest to accommodate early in the design phase

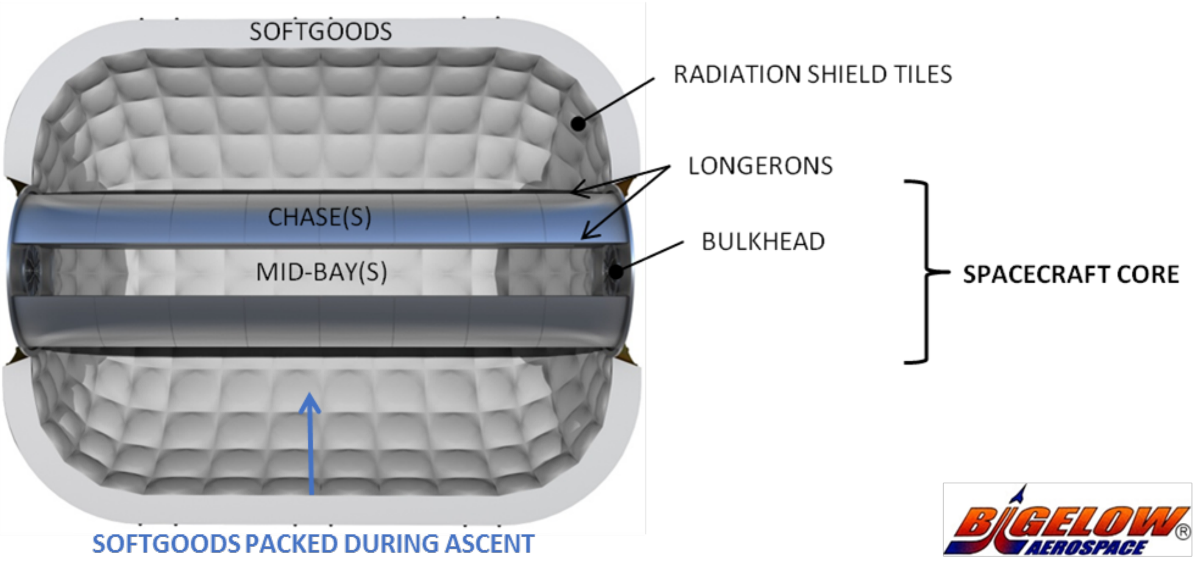


Figure 3-5: B330 texture.

Note: Internal volume exposed to launch ascent decompression and vacuum before expansion. Equipment transported via the crew capsule can move into the B330 volume through the hatch.

- Spacecraft Core (Chases and Mid-Bays)
  - Sustained access to vacuum and fluid lines near core structure
  - Payload racks can be built into the chases
  - BA has developed a standardized seat track mounting system
- Internal Air Barrier Surface (Post-Expansion)
  - Crew stations, science experiments and hardware requiring basic power and data connection may be moved to the internal air barrier surface

Crew and equipment will enter the B330 through the hatch and into the core, between the chases and mid-bays. Paths through the mid-bays, at least 1 m wide, allow translation toward the wall of the structure (also referred to as the internal air barrier).

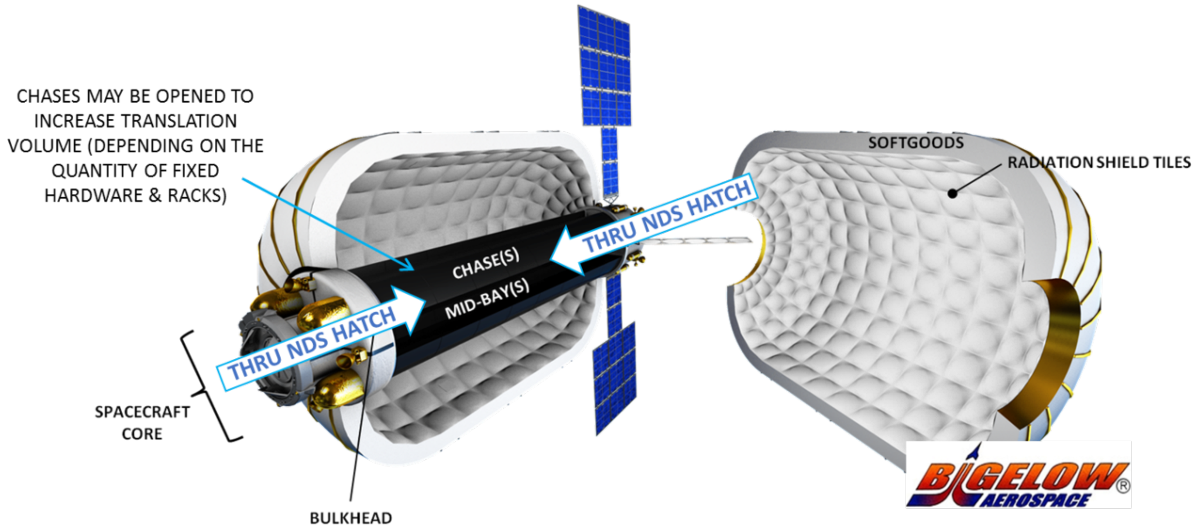


Figure 3-6: B330 access.

In contrast to the solid space station modules, where items or racks are installed on the module's wall, in the Bigelow module everything must be mounted to the central structure during launch. That leads to a completely different usage of space and astronaut movements / orientation.

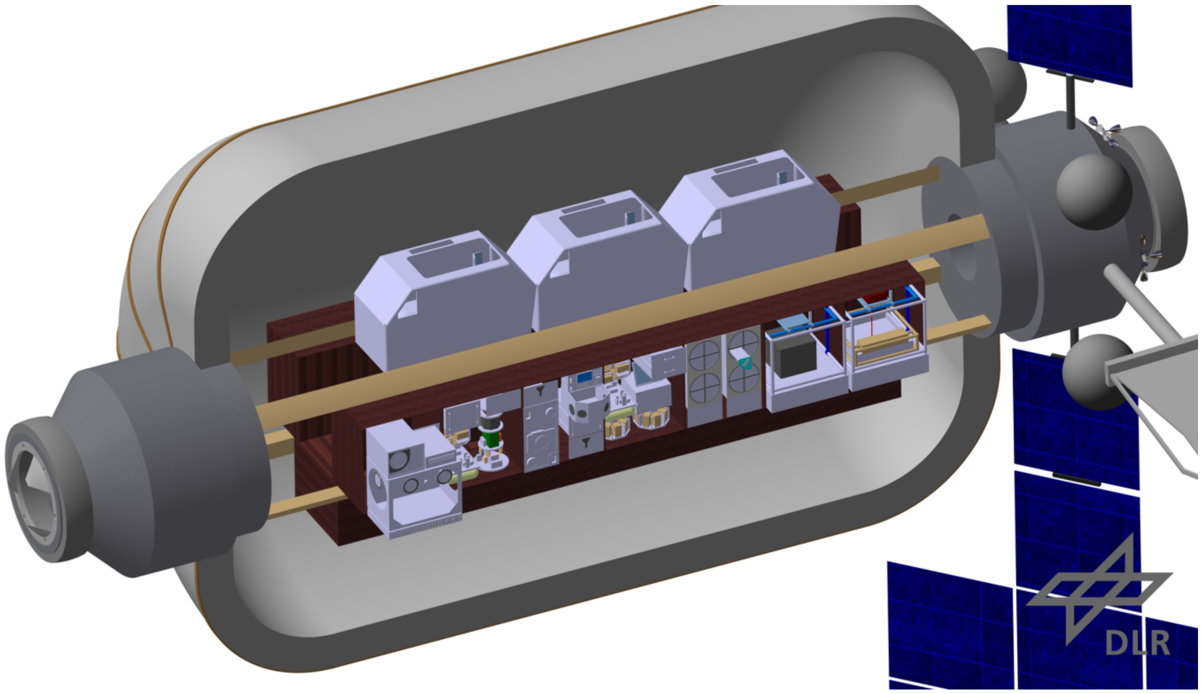
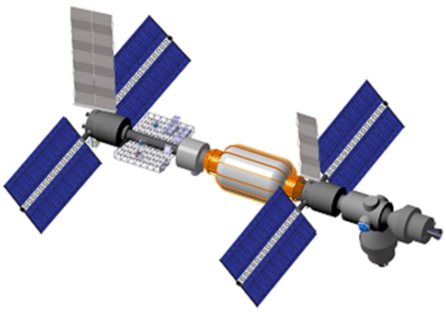
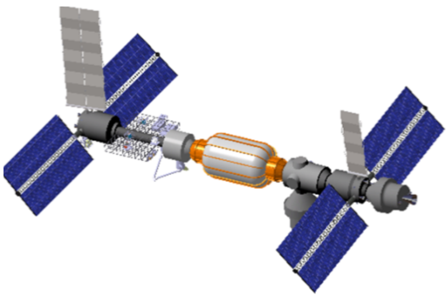
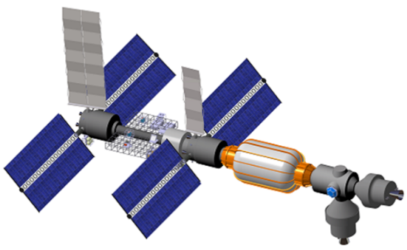


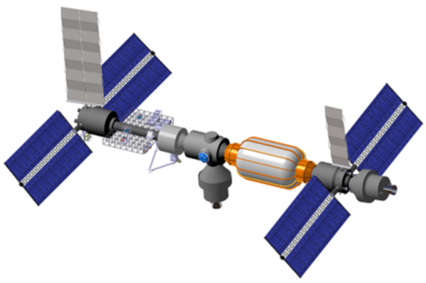
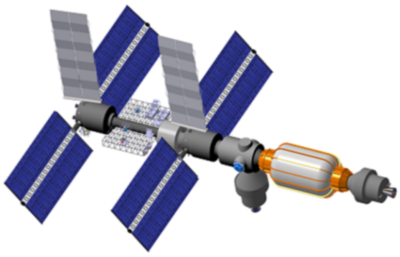
Figure 3-7: Habitat module concept during Post-ISS Scenario-I study.

3.3. Options and Trades

During the study the position of the single sub-modules has been discussed. Advantages and disadvantages of the possible options can be seen in Table 3-1. Option 1 was finally the most favourite one, mainly for docking clearance reasons. Still the Free-Flyer would be able to dock either to the habitat or to the docking node.

Table 3-1: Options of various module positions.

Option	Advantages	Disadvantages
1 	• airlock (on B330) close to platform • clearance (one half sphere) at docking node and habitat module	• no re-boost of base station with Service Module (but with thrusters on node and visiting vehicle, which is attached to the station during all crewed time anyway)
2 	• re-boost (AOCS) possible by Service module • airlock close to platform	• little clearance for docking operations due to solar array, radiators and habitat module
3 	• service module could re-boost in separated mode • free clearance for docking operations at node site	• airlock is very distanced to platform, robotic arm would have to move past solar array/ radiators, due to diameter change of modules no rail could be used for robotic arm

4		• more clearance for docking node • re-boost possible by Service module	• airlock further from platform
5		• re-boost possible by service module	• airlock far from platform/ obstructed transfer way due to solar array and radiators • little clearance for docking operations due to solar array, radiators and habitat module

3.4. To Be Further Studied / Additional Considerations

- A system margin of 20% might not be fully applicable for a space station as used for satellites.

3.5. Summary

In comparison to the ISS the Orbital-Hub concept is much smaller, but would continue similar functions such as permanent housing of a crew (including e.g. human physiology experiments), orbit maintenance, power and thermal supply or flexible docking capacity. In contrast to the ISS EVAs are foreseen only as a contingency. Science and LEO applications can find improved conditions on the man-tended Free-Flyer, which is laid out in more detail in the Post-ISS Scenario II study.

4. Payloads / Science

4.1. Requirements and Design Drivers

- consider proposals from science community (DLR-workshop May 2014 and CEF-study Dec 2014)
- make reasonable assumptions for mass/power/etc-requirements where not specified yet by discipline-experts(→ to be confirmed and updated later)
- include system component "Shared_Equipment_in_Habitat" with several multi-user facilities: Incubator, RefCentrifuge, Freezer, Refrigerator, Glovebox
- include place-holder ("General_Health_Research_Exp"; ~¾ ISPR) for crew health monitoring and future experiments in Human Physiology or Medicine because up to now we have only one proposal as example-experiment "Intracranial_Pressure" (which is ~½ ISPR)
- include general place-holder for "Future_Science" Exp. (~ 1.5 ISPR)
- include "Robotic_Experiments" – proposed by Robotics –

4.2. Baseline Design

4.2.1. Gravitation-Biology

- FLUMIAS (~ ½ ISPR):
  - Modular Laser-Scan-/ Fluorescence Microscope with Centrifuge as reference (1g) → life cell imaging
  - Corresponding Microscope test on TEXUS 2015 (FLUMIAS, Airbus DS)
  - Telemetry (operations from ground)
  - Usage of Incubator/Freezer and Glovebox
    - Mass (with margin): 240 kg
    - Power (avg wMargin): 55.68 W
    - Max. Power (wMargin): 180 W

4.2.2. Human-Physiology

- Intracranial_Pressure (~ ½ ISPR)
  - Measurements: pressure (Vittamed), ultrasound and Near-Infrared-Spectroscopy (NIRS); blood samples
  - Tele-communication/-presence for ultrasound investigations

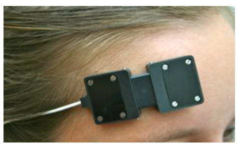


Figure 4-1: Example Equipment for intracranial pressure experiments

- Usage of Centrifuge, Freezer and Glovebox
    - Mass (with margin): 240 kg
    - Power (avg wMargin): 14.376 W
    - Max. Power (wMargin): 1200 W

- General_Health_Research (~ ¾ ISPR)
    - Placeholder for crew health monitoring and future experiments in Human Physiology or Medicine
        - Mass (with margin): 600 kg
        - Power (avg wMargin): 3001.2 W
        - Max. Power (wMargin): 6000 W

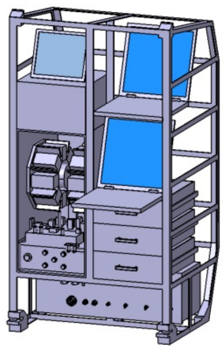


Figure 4-2: Example Equipment for Gravitation Biology and Human Physiology

4.2.3. Robotic-Experiment

- Robot_Assistant (~ 1/6 ISPR)
    - Testing and experimentation (internal)
    - Maintenance & repairs (external)
        - Mass (with margin): 120 kg
        - Power (avg wMargin): 12 W
        - Max. Power (wMargin): 1200 W

- NanoSat_Freeflyers (~ 1/8 ISPR)
    - Platform for formation flight and close-proximity operations
    - Internal and external use
        - Mass (with margin): 60.96 kg
        - Power (avg wMargin): 9 W
        - Max. Power (wMargin): 180 W

Figure 4-3: DLR-Justin (top) and MIT SPHERES (bottom)

(Described in more detail in the robotics section 14.3 , page 108.)

4.2.4. Shared-Equipment

- Incubator & RefZentrifuge (~ ¼ ISPR)
  - Mass (with margin): 120 kg
  - Power (avg wMargin): 62.16 W
  - Max. Power (wMargin): 600 W

- Freezer & Refrigerator (~ ¼ ISPR)
  - Mass (with margin): 120 kg
  - Power (avg wMargin): 600 W
  - Max. Power (wMargin): 600 W

- Glovebox (~ ¼ ISPR)
  - Mass (with margin): 60 kg
  - Power (avg wMargin): 37.68 W
  - Max. Power (wMargin): 120 W

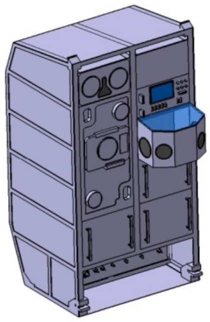


Figure 4-4: Example rack with Glovebox

4.2.5. Radiation-Biology

- Biology_Insitu_Radiation_Damage (~ 1/10 ISPR)
  - Radiation plus microgravity effects on biological cells
  - Standardized experiment containers
  - CELLRAD hardware

  - Mass (with margin): 24 kg
  - Power (avg wMargin): 19.68 W
  - Max. Power (wMargin): 24 W

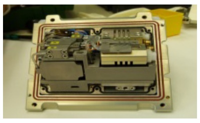


Figure 4-5: Insitu radiation experiment

- Radiation_Dosimetry_Phantomexperiment (~ ¼ ISPR)
  - Phantom module with active/passive individual dosimeters (exposure outside/inside HAB)

  - Mass (with margin): 120 kg
  - Power (avg wMargin): 31.2 W
  - Max. Power (wMargin): 60 W

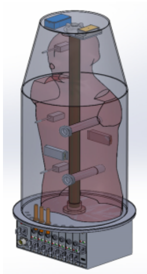


Figure 4-6: Phantom experiment

- Gen. Radiation Dosimetry Habitat (~ 1/10 ISPR)
    - Local and personal radiation dosimetry
    - Stationary active dosimeter unit plus 20 small units distributed inside the HAB-module
    - Active/passive personal dosimeter units (~10)

        - Mass (with margin): 36 kg
        - Power (avg wMargin): 48.48 W
        - Max. Power (wMargin): 60 W

Figure 4-7: Radiation measurement experiments

4.2.6. Technology-Demonstration

- Manufacturing_Workbench (~ 1/2 ISPR)
    - 3D-Printing, milling / laser cutting

        - Mass (with margin): 360 kg
        - Power (avg wMargin): 481.92 W
        - Max. Power (wMargin): 2400 W

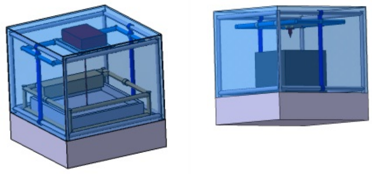


Figure 4-8: 3-D-printer boxes

4.2.7. Future Science

- Placeholder (~ 1.5 ISPR)
    - Significant yet undefined spare capacity reserved inside pressurized volume due to long lead time of investigation

        - Mass (with margin): 1200 kg
        - Power (avg wMargin): 3601.2 W
        - Max. Power (wMargin): 7200 W

4.3. Payload Budgets

- total: 3300 kg corresponding to ~ 5 ISPRs

4.3.1. List of Equipment

Table 4-1: Mass budget of the payloads.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Future_Science	1000.00	20.00	200.00	1200.00	36.35
Gravitationsbiologie	200.00	20.00	40.00	240.00	7.27
Humanphysiologie_Medizin	700.00	20.00	140.00	840.00	25.45
Robotic_Experiments	150.80	20.00	30.16	180.96	5.48
Shared_Equipment_in_Habitat	250.00	20.00	50.00	300.00	9.09
Strahlenbiologie	150.00	20.00	30.00	180.00	5.45
Technologiedemo_Exploration	300.00	20.00	60.00	360.00	10.91

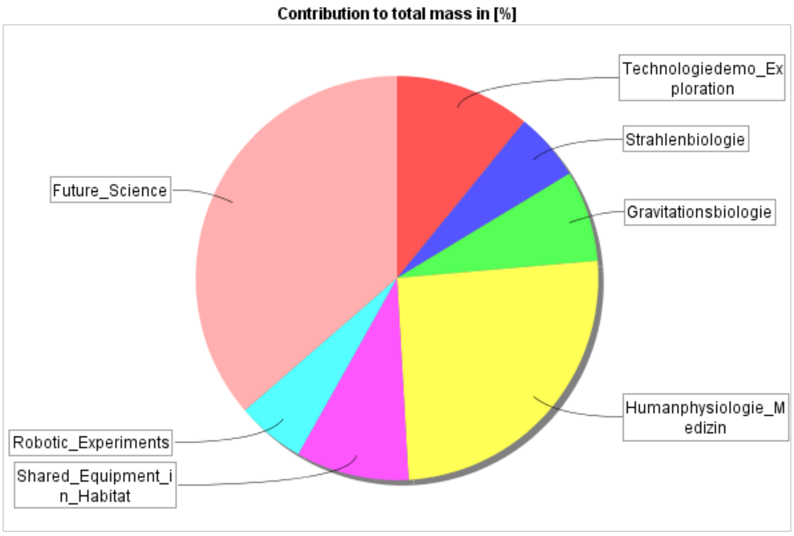


Figure 4-9: Pie chart of mass distribution of all payloads

4.3.2. Power Budget

Maximum total power requirements (all experiments on in parallel) are quite high: ~ 20 kW.

However most experiments have duty cycles between 5 and 50 %, so that suitable experiment scheduling and time sequencing should be possible with an average power demand of up to ~ 8 kW.

Table 4-2: Power budget of the payloads.

Science Power Requirements Summary

Science_on_BaseStation	power_avg_wMargin	Watt	7974.576	761.520
Gravitationsbiologie	power_avg_wMargin	Watt	55.680	0.000
Humanphysiologie_Medizin	power_avg_wMargin	Watt	3015.576	0.000
Strahlenbiologie	power_avg_wMargin	Watt	99.360	99.360
Technologiedemo_Exploratio	power_avg_wMargin	Watt	481.920	0.000
Shared_Equipment_in_Habitat	power_avg_wMargin	Watt	699.840	662.160
Robotic_Experiments	power_avg_wMargin	Watt	21.000	0.000
Future_Science	power_avg_wMargin	Watt	3601.200	0.000
			total: 7975 Watt	Survival Mode

4.3.3. Mode Dependencies

For the performance of the science experiments no significant dependence on the different power modes is identifiable, except for the survival mode, when most science experiments must be shut down. If Base Station re-boost is done with electric thrusters (which requires about 10 kW) supply of power for experiments in the Standard Mode (electrical) may be short, depending on where the power for the thrusters comes from (e.g. Base Station?).

4.3.4. Data Rate & Volume

Required data rates and volumes for scientific payloads are moderate, considering the low duty cycles in most cases proposed. Some video links and teleoperations/telepresence from ground are proposed (e.g. in Gravitation Biology and Human Physiology), but they are not excessive and stay within ISS-standards.

4.4. Re-supply Items / Return Capability

- Gravitation Biology
  Upload and download of frozen samples: few kg per half year
- Human Physiology:
  12 astronauts desirable per experiment with a variety of experiments
  Resupply: ~ 5 kg per astronaut per exp. (expendable items)
  Return Capability: 300 g frozen samples per astronaut per experiment
- NanoSatFreeflyers
  Tanks (~200g, 50 per year)
  Batteries (~100g, 50 per year), alternative: use accumulators

- Radiation Biology
  Exchange of passive detectors ~ few kg (every 6-12 month)
  Resupply and return of frozen biological samples ~ few kg per half year
- Manufacturing Workbench
  Resupply of material for 3D-printing ~ few kg per month
  Return Capability: ~ few kg per month for ground inspection/tests

4.5. To Be Further Studied / Additional Considerations

- Check and confirm or modify numbers with experts
- Fill placeholders with reasonable ideas and numbers

4.6. Summary

The up to now proposed (DLR internal) scientific (strawman-) payloads for the Base Station are mostly derived/extrapolated from present ISS-research and require only moderate resources. They do not call for any special novel support from a future LEO space station and thus represent no significant design driver for a manned Post-ISS infrastructure from the science point of view. Commercial utilization proposals are not considered at the moment, but may be little anyway. A broader poll within the scientific community and industry in Germany and Europe could extend the basis of information and is strongly recommended.

5. Crew Facilities

5.1. Requirements and Design Drivers

- Keeping crew comfortable and in excellent mental and physical health
- Keeping crew at an excellent state of physical fitness
- Each crewmember needs a limited personal space of privacy

5.2. Baseline Design

- Sleepstations, Foodstation_Kitchen and Hygiene_Station located close together in HA to create a Crew Quarters atmosphere
- Toilet_Urine_Feces located away from Sleepstations in SM for noise reasons
- Treatmill and ARED located in DN for noise reasons and due to availability of space

5.3. Mass and Power Budget

- Rough estimate
- High potential of improved designs

5.3.1. List of Equipment

Food not included

Table 5-1: Mass budget of the crew facilities.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
DN_ARED	200.00	10.00	20.00	220.00	15.20
DN_Treatmill	200.00	10.00	20.00	220.00	15.20
HA_Clothing	364.00	5.00	18.20	382.20	26.41
HA_Crew_Preference_Items	30.00	20.00	6.00	36.00	2.49
HA_Foodstation_Kitchen	70.00	20.00	14.00	84.00	5.80
HA_Hygiene_Station_Items	30.00	10.00	3.00	33.00	2.28
HA_Pharmacy	10.00	5.00	0.50	10.50	0.73
HA_Sleepstations	240.00	10.00	24.00	264.00	18.24
HA_Trash_Gathering_Removal	16.00	10.00	1.60	17.60	1.22
SM_Toilet_Urine_Feces	150.00	20.00	30.00	180.00	12.44

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]
Total dry mass:	1310.00			1447.30

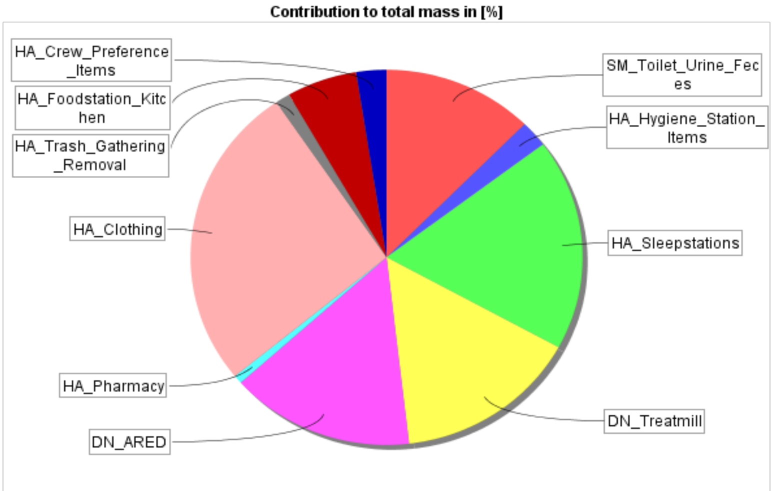


Figure 5-1: Pie chart of mass distribution of the crew facilities

5.3.2. Power Budget

Table 5-2: Power budget of the crew facilities.

System Component	Parameter	Unit	Default	Standard_elect...	CrewExchange	Docked	Survival	Proximity_Ops
CrewFacilities	power_avg_wMargin	Watt	320.240	320.240	320.240	320.240	179.440	320.240
DN_ARED	power_avg_wMargin	Watt	39.600	39.600	39.600	39.600	0.000	39.600
HA_Clothing	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
HA_Crew_Preference_Items	power_avg_wMargin	Watt	9.600	9.600	9.600	9.600	9.600	9.600
HA_Foodstation_Kitchen	power_avg_wMargin	Watt	112.080	112.080	112.080	112.080	112.080	112.080
HA_Hygiene_Station_Items	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
HA_Pharmacy	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
HA_Sleepstations	power_avg_wMargin	Watt	35.200	35.200	35.200	35.200	35.200	35.200
SM_Toilet_Urine_Feces	power_avg_wMargin	Watt	22.560	22.560	22.560	22.560	22.560	22.560
HA_Trash_Gathering_Remova	power_avg_wMargin	Watt	2.200	2.200	2.200	2.200	0.000	2.200
DN_Treatmill	power_avg_wMargin	Watt	99.000	99.000	99.000	99.000	0.000	99.000

5.3.3. Mode Dependencies

Low influence of various modes with the exception of Survival Mode, in which exercise most likely will be dropped.

5.4. Re-Supply Items

- Food incl. Trashbags, with every docking vehicle, minimum1 per half year
- Hygiene items, with every docking vehicle, minimum1 per half year
- Toilett supplies, with every docking vehicle, minimum1 per half year
- Clothing for new crew, every 180 days
- Crew items for new crew, every 180 days

5.5. Summary

- Crew Facilities are of major importance for ' Long Duration Missions'
- They have a lot of potential for design improvement, because they are taken more seriously now, since 'Long Duration Missions' are the nominal case.

6. ECLSS

6.1. Requirements and Design Drivers

- Secure a comfortable and secure work and life environment for three Astronauts
- For limited time (up to 1 week), provide the same for 6 Astronauts (during direct Handover)

6.2. Baseline Design

- Due to the essential nature of ECLSS we plan three identical instruments, one in each pressurized modules of BaseStation, serving also as Backup
- For normal Ops, one system is used all the time and one half the time
- For direct Handover we use all three systems
- For Survival Mode we use one system only
- O_2 generation,CO_2 removal and Waterrecycling use the newest versions, as used recently on the US side of ISS

6.3. Options and Trades

- O_2 Storage can use exchangeable tanks or use a Refill Pump Assembly
- Fire extinguishers can use CO_2, Halon or the Russian water soap mix (TBD)
- Gas (O_2 and N_2) high pressure stowage tanks might need to integrated outside

6.4. Mass and Power Budget

- Rough estimates

6.4.1. List of Equipment

Table 6-1: Mass budget of the ECLSS.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
CO2_Removal	390.00	20.00	78.00	468.00	12.66
Fire_Detection_Suppression	15.00	10.00	1.50	16.50	0.45
H2O_Supply_Recycling	660.00	20.00	132.00	792.00	21.43
Lighting	45.00	10.00	4.50	49.50	1.34
N2_Pressure_Storage	600.00	20.00	120.00	720.00	19.48
O2_Generation_Storage	660.00	20.00	132.00	792.00	21.43
Temperature_Humidity_Control	390.00	10.00	39.00	429.00	11.61
Trace_Contamination_Removal	240.00	10.00	24.00	264.00	7.14
Ventilation	150.00	10.00	15.00	165.00	4.46

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]
Total dry mass:	3150.00			3696.00

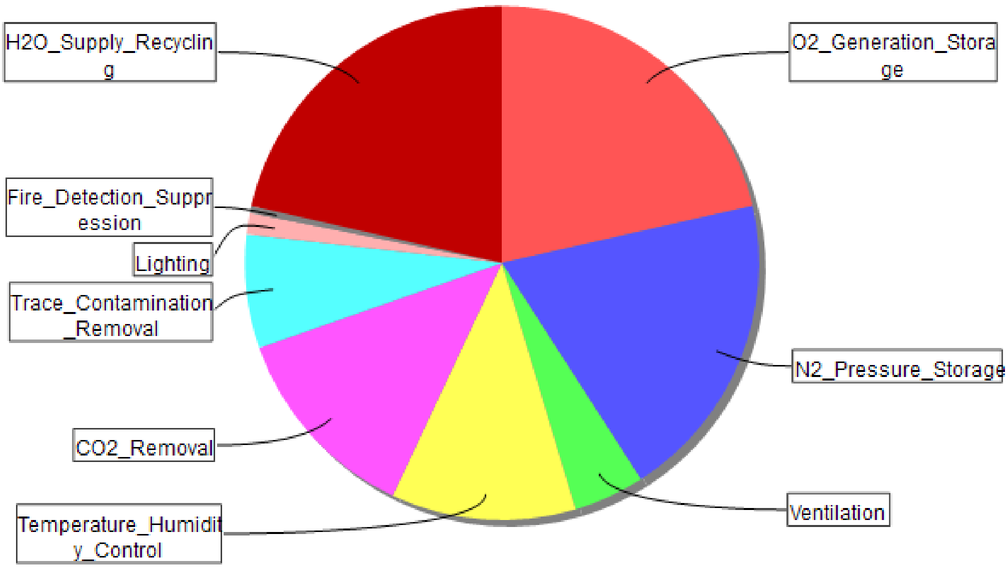


Figure 6-1: Pie chart of mass distribution of the ECLSS

6.4.2. Power Budget

Table 6-2: Power budget of the ECLSS.

System Component	Parameter	Unit	Default	Standard_elect...	CrewExchange	Docked	Survival	Proximity_Ops
ECLSS	power_avg_wMargin	Watt	7092.904	7092.904	9219.840	7092.904	3475.972	7092.904
CO2_Removal	power_avg_wMargin	Watt	318.432	318.432	480.000	318.432	161.616	318.432
Fire_Detection_Suppression	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
H2O_Supply_Recycling	power_avg_wMargin	Watt	1902.432	1902.432	2880.000	1902.432	953.616	1902.432
Lighting	power_avg_wMargin	Watt	212.080	212.080	212.080	212.080	212.080	212.080
N2_Pressure_Storage	power_avg_wMargin	Watt	98.880	98.880	98.880	98.880	98.880	98.880
O2_Generation_Storage	power_avg_wMargin	Watt	3849.600	3849.600	4800.000	3849.600	1473.600	3849.600
Temperature_Humidity_Conti	power_avg_wMargin	Watt	440.880	440.880	440.880	440.880	440.880	440.880
Trace_Contamination_Remov	power_avg_wMargin	Watt	72.600	72.600	110.000	72.600	36.300	72.600
Ventilation	power_avg_wMargin	Watt	198.000	198.000	198.000	198.000	99.000	198.000
ECLSS	power_energy_wMargin	Joule	8579576678.400	8579576678....	5576159232....	4289788339....	1201295923....	612826905.600

6.4.3. Mode dependencies

- The used power depends mainly on the number of crew to be supported.
- 'Crew Exchange' is the most consuming mode when a 'Direct Handover' is executed.
- During 'Survival Mode' the use of ECLSS can be minimised, but only for a limited time. It directly influences crew comfort and crew capabilities.

6.5. Re-Supply Items

- Water to compensate loss, with every visiting vehicle
- Filter exchange for Trace_Contamination_Removal, with every supply vehicle
- O_2, refill tank, exchange High Pressure Tank, as needed (every two years)
- N_2, refill tank, exchange High Pressure Tank, as needed (every three years)

6.6. To Be Further Studied / Additional Considerations

- O_2 High Pressure Pump Assembly

6.7. Summary

Experience with ISS shows that it is essential for a safe and reliable uninterrupted operation to have additional ECLSS capabilities for redundancy and in order to master various 'Off Nominal' situations.

7. EVA

7.1. Requirements and Design Drivers

- Airlock located on HA docking port towards FF
- Limited capabilities of EVA for contingency cases only
- EVA suits for two crew members
- One set of service equipment

7.2. Baseline Design

- Habitat:
  - Two EVA suits
  - Exchange parts for two suits in different sizes
  - Battery charging station
  - CO_2 filter regenerator
  - O_2 high pressure tank + pump (outside of HA located near airlock for safety)
  - Tools and tethers for EVA

7.3. Options and Trades

- Option 1: O_2 high pressure tank + pump located inside of HA
- Option 2: No O_2 high pressure pump → exchange of tanks

- EVA for installation purposes reduced to absolute minimum

7.4. Mass and Power Budget

Experience on ISS shows that the lifetime of the unmaintained EMU on Orbit has to be reconsidered. It will be shorter than the assumed 6 years. Which will lead to increased 'Mass Budget'.

7.4.1. List of Equipment

Table 7-1: Mass budget of the EVA equipment.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
HA_Battery_Charge_Station	30.00	10.00	3.00	33.00	4.03
HA_CO2_Filter_Regeneration	20.00	10.00	2.00	22.00	2.69
HA_EVA_Suits_2	280.00	10.00	28.00	308.00	37.65
HA_O2_High_Pressure_pump	100.00	20.00	20.00	120.00	14.67
HA_O2_High_Pressure_Tank	200.00	10.00	20.00	220.00	26.89
HA_Resize_Exchange_Parts	50.00	10.00	5.00	55.00	6.72
HA_Tools_Tethers	50.00	20.00	10.00	60.00	7.33

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	730.00			818.00	

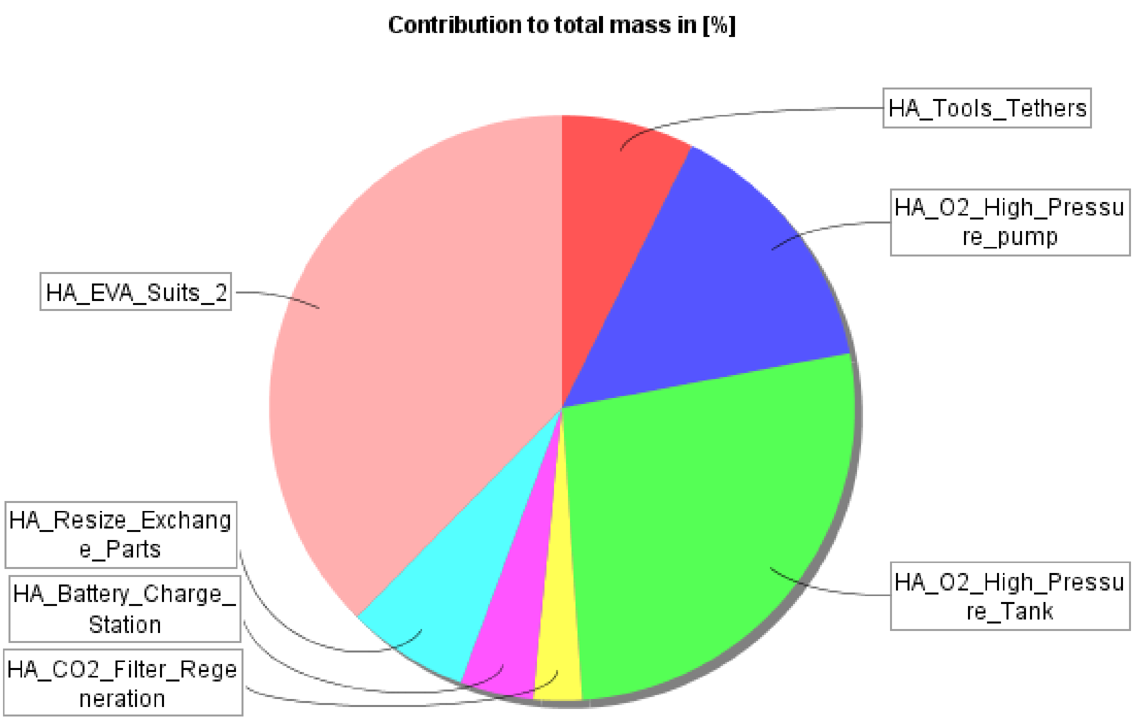


Figure 7-1: Pie chart of mass distribution of the EVA equipment

7.4.2. Power Budget

No relevant power demand due to low duty cycle.

7.4.3. Mode dependencies

- No direct mode dependencies.
- The optimum is to execute an EVA out of a well-rested state with prior thorough planning and preparation.

7.5. Re-Supply Items

- Suits (2), (pre ‚Water in the Helmet' event, every 6 years),
  (now, every 4 years, TBD)
- O_2 in High Pressure Tanks (in case of no ‚O_2 High Pressure Pump Assembly' on HA)

7.6. To Be Further Studied / Additional Considerations

Is an "O_2 High Pressure Pump Assembly" realizable for use in Space?

7.7. Summary

- EVA capability might be essential for 'Off Nominal' Situations'
- EVA capability will be needed for the integration phase, for planned and unplanned tasks.

8. Mission Analysis

8.1. Requirements and Design Drivers

- Target orbit: Inclination 51.6°; Altitude=400 km
- DryMass=50000 kg; A_max = 500 m²
- Solar Flux Fp=150…300

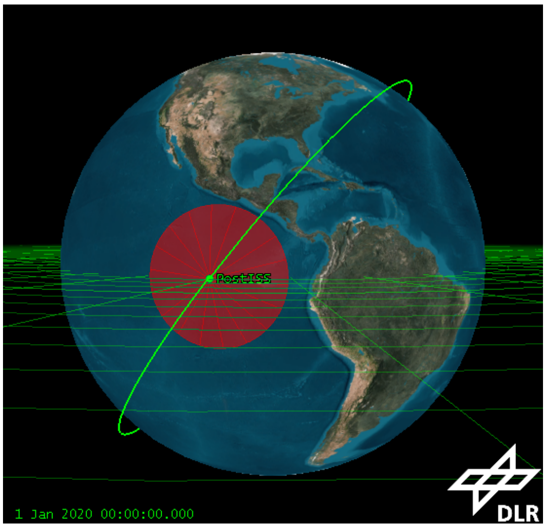


Figure 8-1: Post-ISS orbit visualization

8.2. Baseline Design

The baseline orbit design is correspondent to the requirements. Since there is no strict nadir pointing required anymore for the Base Station, it is free to yaw or roll around its longitudinal axis in order to point the one-axis-movable solar panels to the Sun (according to the beta-angle).

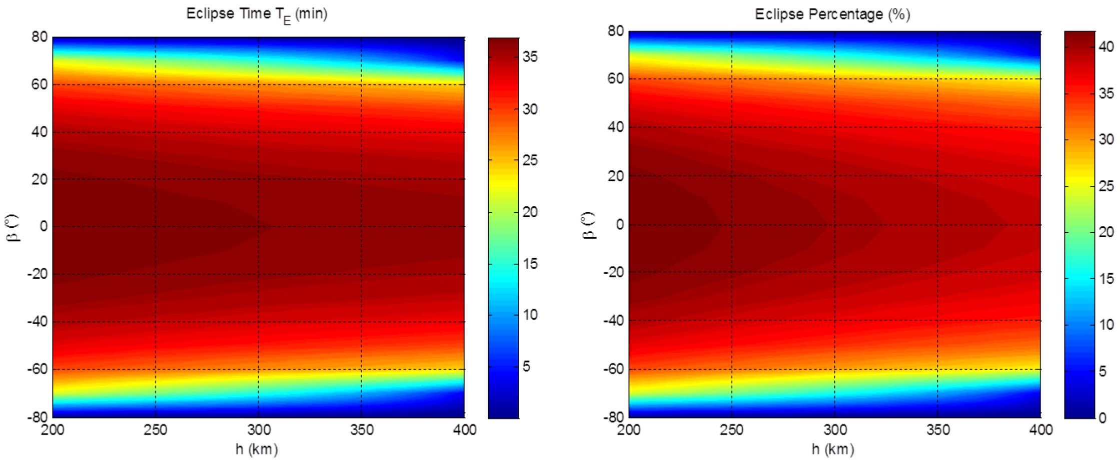


Figure 8-2: Eclipse times depending on beta-angle and altitude (left: in min.; right: in %)

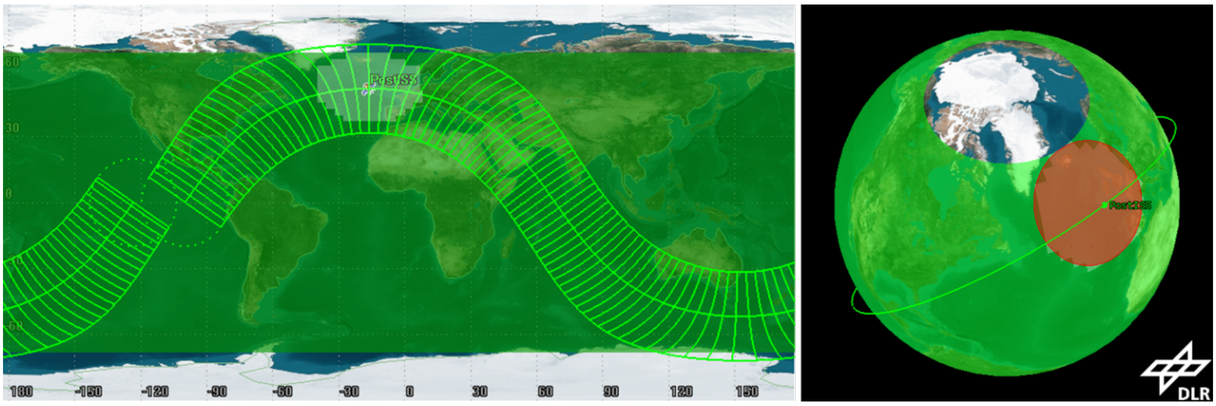


Figure 8-3: Coverage of the baseline orbit (51.6°incl.; 5° elevation)

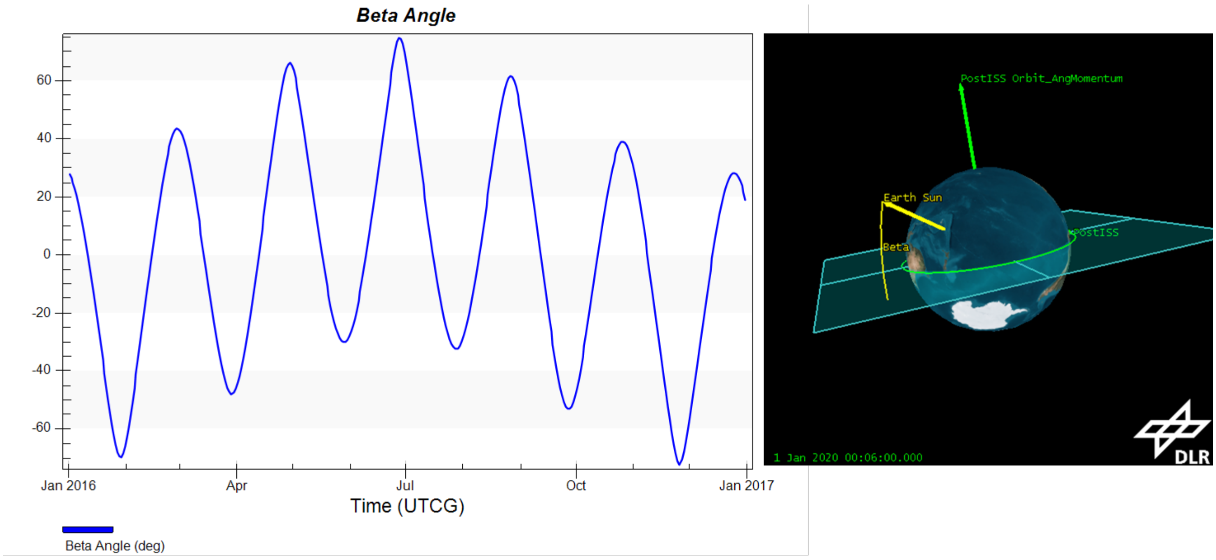


Figure 8-4: Beta-angle of the baseline orbit in deg. over one year.

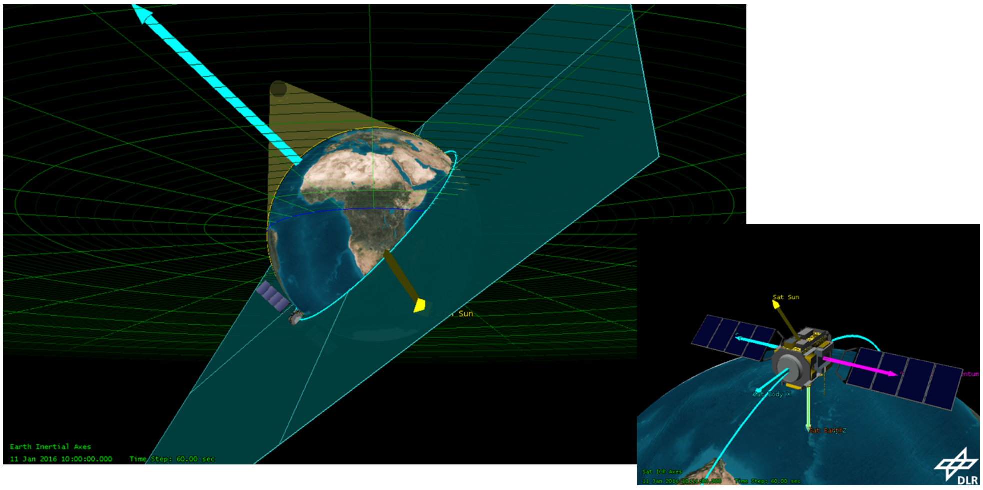


Figure 8-5: Baseline attitude when beta-angle is 0°.

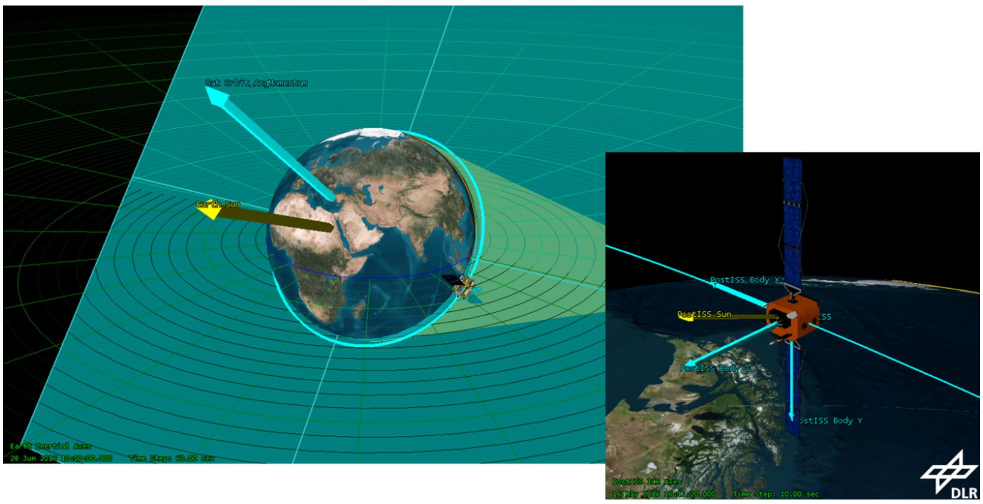


Figure 8-6: Baseline attitude when beta-angle is 75°.

8.3. Options and Trades

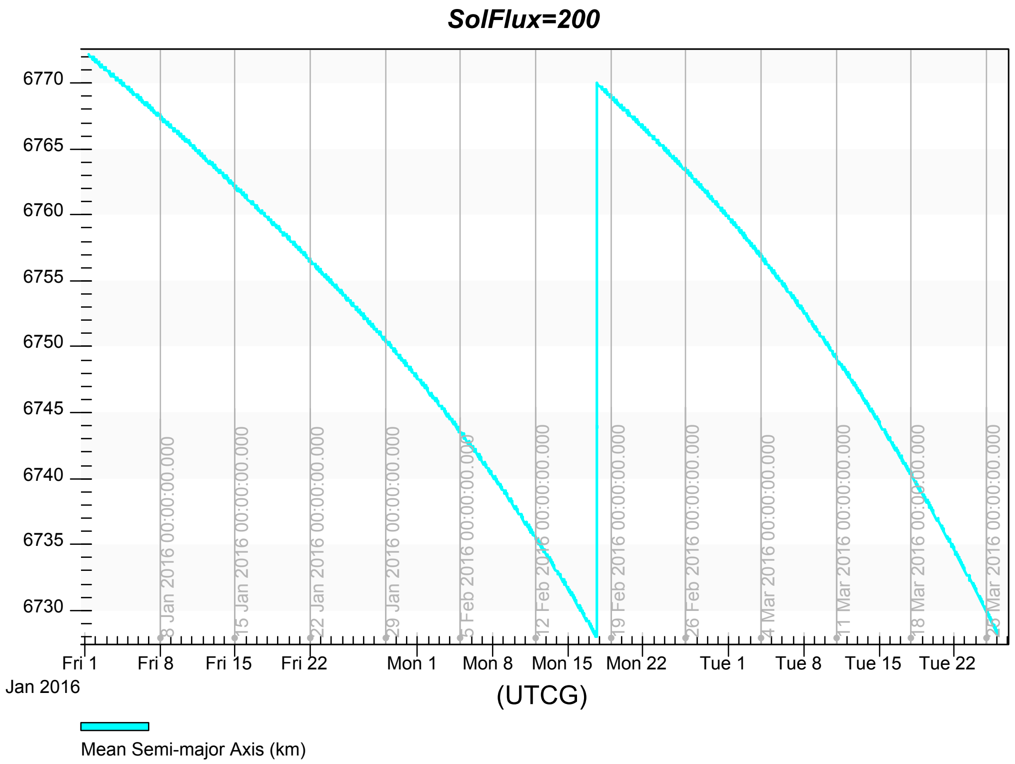


Figure 8-7: Orbit maintenance behaviour for impulsive manoeuvres.

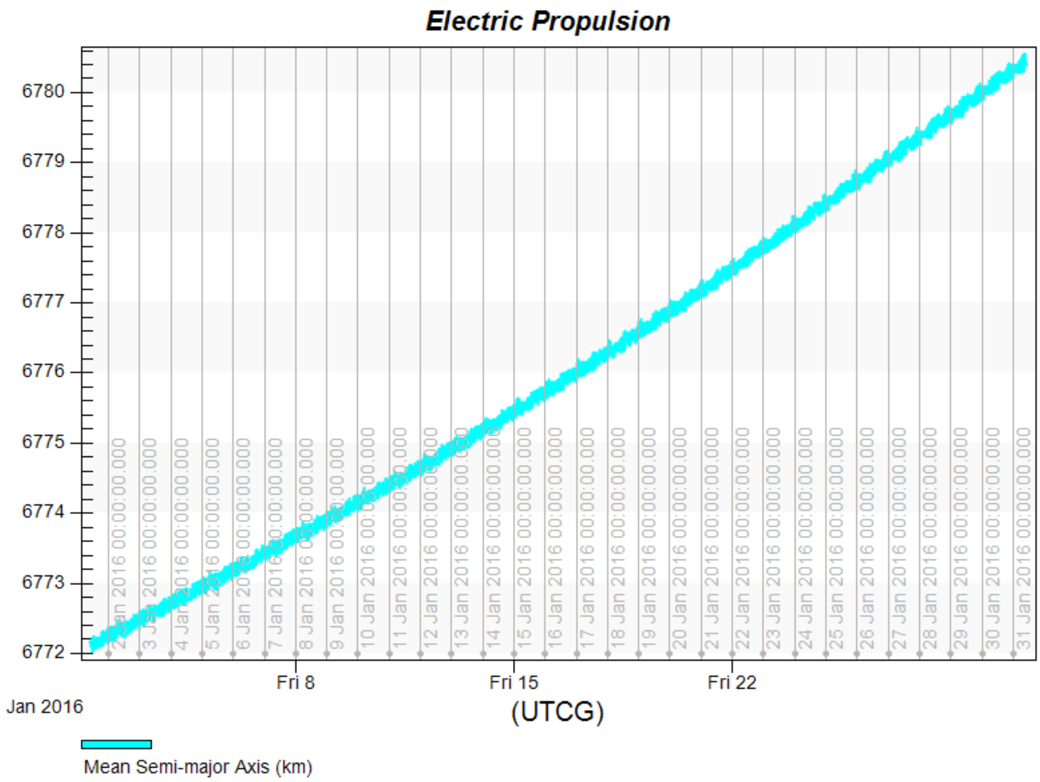


Figure 8-8: Orbit maintenance behaviour for electric propulsion.

Table 8-1: Electric vs. chemical propulsion.

electric			chemical	
RIT-22	0,15	N		
2 x	4,5	kW		
	4400	sec	300	sec
	0,6	kg/day	148	kg / mo
			12	mo / year
	219,2	kg / year	1776	kg / year
	~300	kg/SA		

8.4. To Be Further Studied / Additional Considerations

- Manuever Coupling between FreeFlyer
- Optimal Engine Configuration (Powerconsumption versus I_sp)

9. On-Board Computer and Data Handling

9.1. Assumptions

At this early stage no specific requirements were provided to define any processing requisites. Since the processing capability required by the experiments was not greater than the one in the current ISS, the sole assumption considered was to maintain –at a minimum– the current ISS capabilities.

9.2. Baseline Design

The baseline design proposed as an initial stand point is inherited from the current ISS scheme, providing us with a proven concept.
The ISS counts with 2 differentiated processing elements:

- Station Control Module: 15 enhanced MDMs and 31 Standard MDMs
- Payload Processing Unit(s): 40 laptop computers

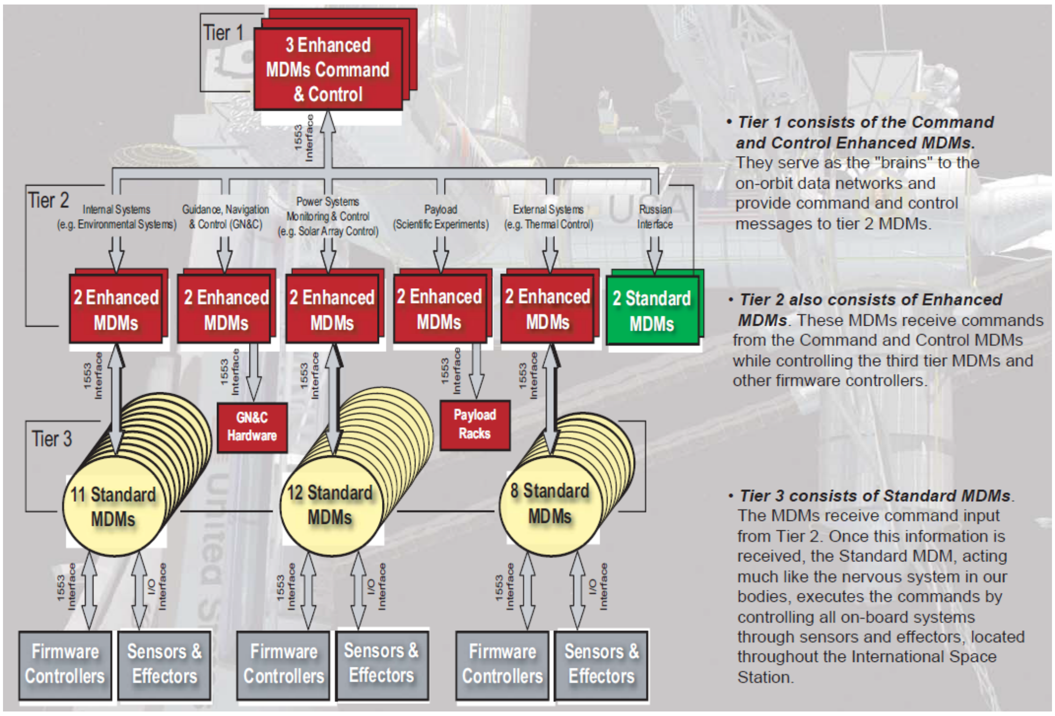


Figure 9-1: ISS MDM Architecture.

9.3. Options and Trades

No information has been found as to whether the computers that are grouped as Payload Processing Units are fully independent, arranged in a similar master-slave configuration as the Station Control Module, or made to work as a cluster. Since individual experiments seem to be dealt with separately, the first case would seem most probable.

As a possible area of optimization, forming a cluster should be considered, as many computers will probably be under-utilized, and a cluster-focused approach, with independent terminals as needed, would probably require less computers and provide better processing capabilities.

Another interesting concept might be to use a cluster of easily changeable computers instead of the MDM's, which in the current station are of difficult access, and have to mostly be repaired, rather than replaced.

9.4. Mass and Power Budget

9.4.1. List of Equipment

Initial estimations, considering the baseline design, were based on Honeywell's definition of the ISS MDM architecture, as well as the use of standard laptops, accounting for some modifications.

Table 9-1: Mass budget of the OBC equipment.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
HA_Payload_Processing_Units	144.00	20.00	28.80	172.80	14.63
SM_Station_Control_Module_EnhMD	465.00	20.00	93.00	558.00	47.26
SM_Station_Control_Module_StdMDI	375.00	20.00	75.00	450.00	38.11

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	984.00			1180.80	

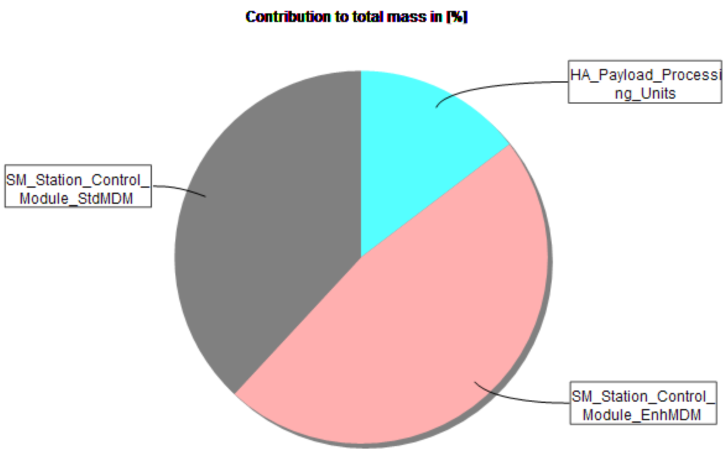


Figure 9-2: Pie chart of mass distribution of the OBC equipment

9.4.2. Power Budget

Table 9-2: Power budget of the OBC system.

System Component	Parameter	Unit	Default	Standard_elect...	CrewExchange	Docked	Survival	Proximity_Ops
OnBoardComputer	power_avg_wMargin	Watt	4442.400	4442.400	4442.400	4442.400	2810.400	4442.400
HA_Payload_Processing_Units	power_avg_wMargin	Watt	3264.000	3264.000	3264.000	3264.000	1632.000	3264.000
SM_Station_Control_Module_	power_avg_wMargin	Watt	540.000	540.000	540.000	540.000	540.000	540.000
SM_Station_Control_Module_	power_avg_wMargin	Watt	638.400	638.400	638.400	638.400	638.400	638.400
OnBoardComputer	power_energy_wMargin	Joule	5373527040.000	5373527040....	2686763520....	2686763520....	971274240.000	383823360.000

9.4.3. Mode dependencies

There is no foreseen difference through the different power modes, as it was established that the whole subsystem should be fully active even in safe mode.

9.5. Re-Supply Items

- Station control module (MDMs): lifetime 8-15 years (depending on installation, might have to be repaired, not exchanged)
- Payload processing units: lifetime 4-7 years

9.6. To Be Further Studied / Additional Considerations

- Clear definition of requirements on the processing side (OBC) and on the communications side (i.e. data rate requirements)
- Memory sizing to be defined according to processing and communications final scheme

9.7. Summary

As no specific requirements were provided to define any processing requisites, the sole assumption considered was to maintain –at a minimum– the current ISS capabilities.
A similar architecture to the one used in the ISS is suggested, with a Station Control Module composed of 46 MDM's that support the station, and an additional 40 laptop computers that conform the Payload Processing Unit which support the experiments.

10. Communication and Ground Segment

10.1. Assumptions

At this early stage no specific requirements were provided to define any data transmission requisites. Since the communication capability required by the experiments was not greater than the one in the current ISS, and no specific increase in video channels or other communication-demanding services has been confirmed, the assumption considered for the communications subsystem was to maintain –at a minimum– the current ISS capabilities, and take note of any communication advances that could be available.

The other assumption relating to communications was the use of both ground stations and GEO data relay systems, one as the main communication channel and the other as a backup.

10.2. Baseline Design

Whether the main communications are channelled through ground stations or through data relay systems, the baseline design will be based on:

- 3 independent systems with exclusive frequency bands, depending on purpose:
    - S-Band: Command, Telemetry, Audio Channels
    - K-Band: Payload, Video data, others
    - UHF (2 independent systems): Extra Vehicular Activities, Docking
- Optional: use of optical communications for high data rates
- Antennas: 1 K-band, 2 S-band (including backup-system), 4 UHF (redundancy)
- Transponders: 2 K-band, 2 S-band, 4 UHF (all include redundancy)
- Note that internal communications are not considered.

As reference, note the current status of the ISS communications Ku-Band:

- Current Ku (updated 2013):
    - Uplink: 25 Mbps (pre-2013, 3 Mbps)
    - Downlink: 300 Mbps (pre-2013, 150 Mbps)

As an option, we foresee the use of optical communications for high data rates. These will be available both for LEO to GEO communications, as well as LEO to Ground communications as follows:

- EDRS (planned launches between 2015 & 2019): Includes a LEO to GEO intersatellite laser link at 1,8 Gbps (tested on ESA's Alphasat I in 2014), scalable to 7,2 Gbps (according to ESA)

- OPALS (experiment currently on ISS): LEO to ground optical comms tested at 50Mbps (2014)

With these base considerations, two cases were explored:

- Case A: Communication channelled through TDRS/EDRS (and Ground Segment planned as a backup)
- Case B: Communication channelled through Ground Segment (and TDRS/EDRS planned as a backup)

Case A: Communication channelled through TDRS/EDRS (inherited from ISS NASA concept)

- Advantages:
    - Proven concept
    - High contact time (maximum period without comms 15 minutes)
    - EDRS will provide optical comms with data rate 1,8 – 7,2 Gbps
    - Using Ka band (not Ku) provides 800 Mbps (vs. 300 Mbps)
- Disadvantages:
    - Requires high stability (precise pointing to GEO)
    - Rolling motions must be compensated through steering (limited max. roll and speed)
    - Real time comms suffer delay due to loop through GEO
- Consider ground S-band comms as backup (independent communication system), and probably K band for video support

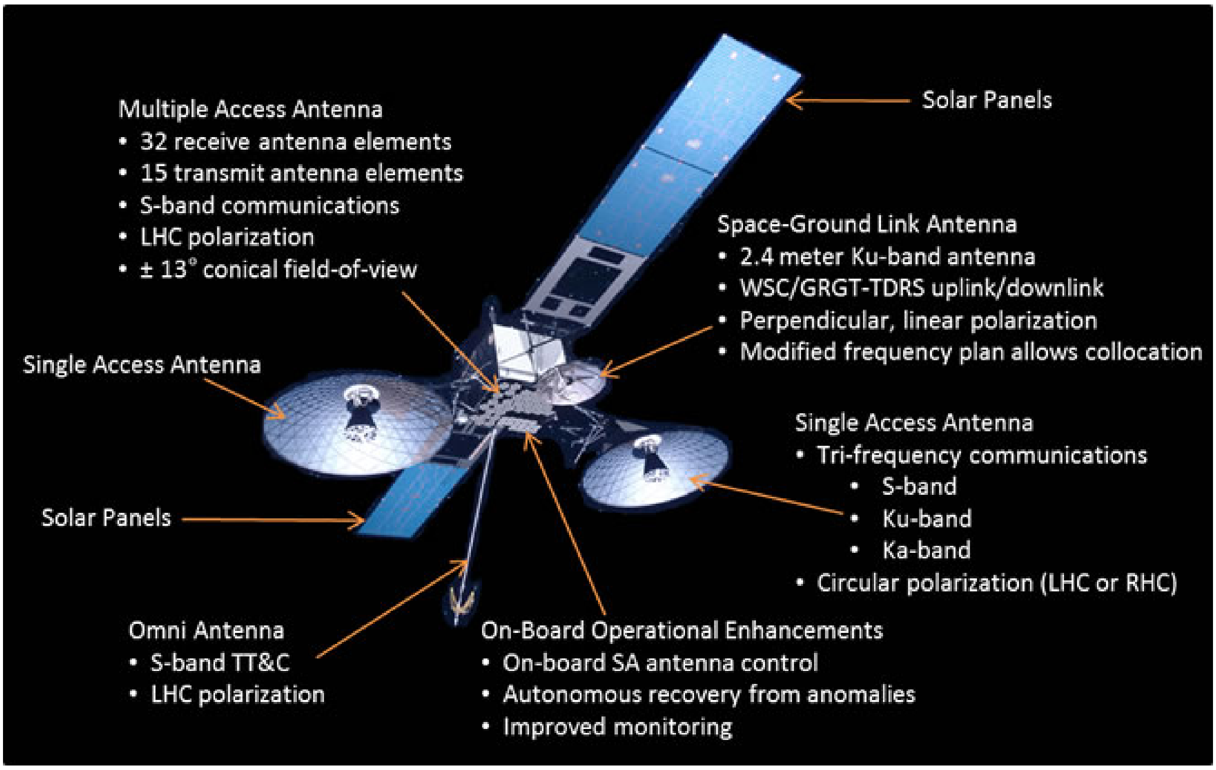


Figure 10-1: NASA TDRS (second generation).

Case B: Communication channelled through Ground Segment

- Advantages:
    - Reduced stability requirement (compared to Case A)
    - Lower transmission power requirements, and complexity
    - Possibly lower cost (tbc)
- Disadvantages:
    - Increased operational complexity
    - Reduced contact time (compared to Case A), and no uninterrupted communications longer than 100 seconds
    - Optical communications available at lower speed (OPALS, 50 Mbps)
    - Real time comms suffer delay due to loop through GEO
- Consider TDRS/EDRS S-band comms as backup (independent communication system), and probably K band for video support

Figure 10-2: ESATRACK (Ground Segment Distribution).

10.3. Options and Trades

- As mentioned above, Optical communications could be used in both cases
- Optical communications payload could be established on the flyer, and have the main body use it as a relay (communications between main body and flyer to be studied according to relative movement and line of sight)

10.4. Mass and Power Budget

10.4.1. List of Equipment

Initial estimations where mostly taken from COTS and supplier datasheets, but a 20% margin was foreseen, due to possible modifications and/or changes in the component characteristics in later phases of design.

Table 10-1: Mass budget of the communication equipment.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
DN_HA_UHF_Antenna	16.00	20.00	3.20	19.20	15.84
DN_K_Band_Antenna	43.00	20.00	8.60	51.60	42.57
DN_Laser_comms	0.00	CalcErr	0.00	0.00	0.00
DN_S_Band_Antenna	8.00	20.00	1.60	9.60	7.92
SM_K_Band_Transponder	12.00	20.00	2.40	14.40	11.88
SM_RF_Distribution_Unit	10.00	20.00	2.00	12.00	9.90
SM_S_Band_Transponder	4.00	20.00	0.80	4.80	3.96
SM_UHF_Transponder	8.00	20.00	1.60	9.60	7.92

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	101.00			121.20	

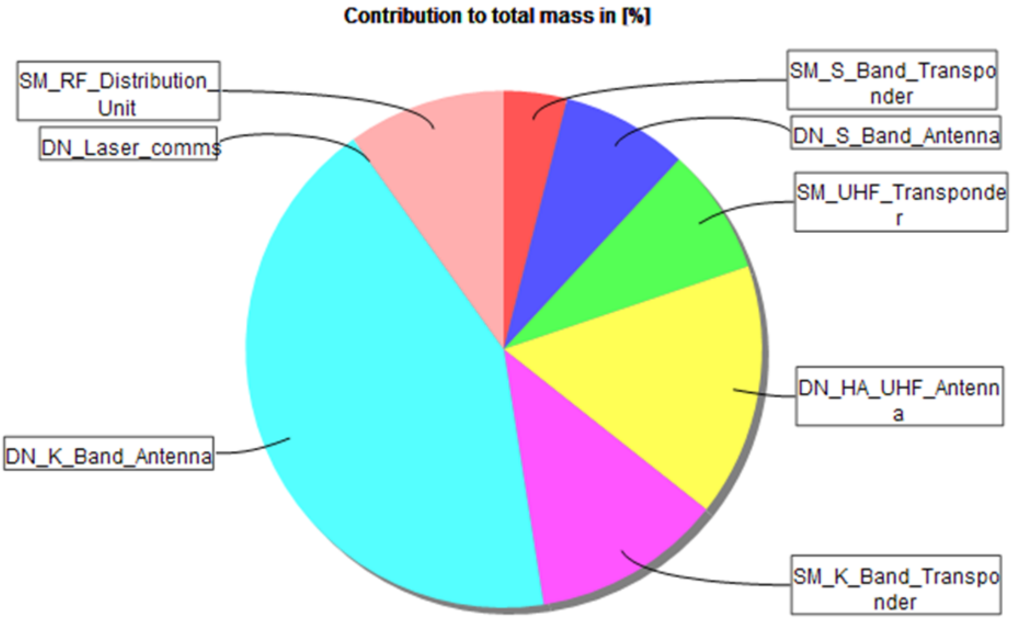


Figure 10-3: Pie chart of mass distribution of the communication equipment

10.4.2. Power Budget

Table 10-2: Power budget of the communication system.

System Component	Parameter	Unit	Default	Standard_elect...	CrewExchange	Docked	Survival	Proximity_Ops
Communication	power_avg_wMargin	Watt	781.200	781.200	781.200	781.200	748.800	781.200
DN_K_Band_Antenna	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
SM_K_Band_Transponder	power_avg_wMargin	Watt	295.200	295.200	295.200	295.200	262.800	295.200
DN_Laser_comms	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
DN_S_Band_Antenna	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
SM_S_Band_Transponder	power_avg_wMargin	Watt	240.000	240.000	240.000	240.000	240.000	240.000
DN_HA_UHF_Antenna	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
SM_UHF_Transponder	power_avg_wMargin	Watt	196.800	196.800	196.800	196.800	196.800	196.800
SM_RF_Distribution_Unit	power_avg_wMargin	Watt	49.200	49.200	49.200	49.200	49.200	49.200
Communication	power_energy_wMargin	Joule	944939520.000	944939520.000	472469760.000	472469760.000	258785280.000	67495680.000

10.4.3. Mode dependencies

There is no foreseen difference through the different power modes, as it was established that the whole subsystem should be fully active even in safe mode: from the point of view of the team, even in safe mode it would be important to maintain video communications, and of course full-duplex voice communications, amongst others.

10.5. Link Budget

Space Station			
Space Station Transmitter Power Output	PTx	16,68	dBW
Space Station Electric Transmitter Power		47	W
Transmission losses between Tx and antenna	LTx	1,50	dB
Antenna Diameter	D	2,00	m
Antenna Efficiency	η	0,50	
Satellite Antenna gain	GTx	46,93	dBi
Minimum Elevation angle	emin	0,00	deg
Space Station EIRP	EIRP	62,11	dBW
Depointing Error	αdep	1,00	deg
Satellite Beamwidth	θ3dB	0,70	deg
Pointing Loss	Ldep	2,45E+01	dB
Downlink Path			
Height Orbiter/Space Station	h	28295786,34	km
Slant range	R	28302163,62	km
Frequency	f	15,00	GHz
Path loss	Lpath	205,00	dB
Isotropic Signal Level at Ground Station		-167,38	dBW
GEO Data Relay Satellite			
Antenna Diameter	D	5,00	m
Antenna Efficiency	η	0,50	
LGA Gain	GRx	54,89	dBi
GEO Data Relay Satellite Losses between antenna and Rx	LRx	0,50	dB
Input Power at S-Band Receiver	PRx	-112,99	dBW
Downlink Temperature	T	34,21	K
		15,34	dBK
Figure of Merit	G/T	39,55	
Orbiter Signal-to-Noise Power Density	C/No	100,27	dBHz
System Desired Data Rate	R	3,00E+08	bps
		84,77	dBHz
Telemetry&Science System Eb/No for the Downlink	Eb/No	15,50	dB
Demodulation Method Selected	BPSK, QPSK		
System Allowed or Specified Bit-Error-Rate	10^-6		
Demodulation Losses	Ldem	1,00	dB
Eb/No Real	Eb/No|real	10,50	dB
System Link Margin		**5,00**	**dB**

Figure 10-4: Link Budget Case A: LEO to GEO link, Ku-Band

Satellite			
Spacecraft Transmitter Power Output	PTx	13,80	dBW
Spacecraft Electric Transmitter Power		24	W
Transmission losses between Tx and antenna	LTx	1,50	dB
Antenna Diameter	D	0,10	m
Antenna Efficiency	η	0,32	
Satellite Antenna gain	GTx	5,00	dBi
Minimum Elevation angle	emin	0,00	deg
Spacecraft EIRP	EIRP	17,30	dBW
Depointing Error	αdep	1,00	deg
Satellite Beamwidth	θ3dB	70,00	deg
Pointing Loss	Ldep	2,45E-03	dB
Downlink Path			
Height Orbiter/Spacecraft	h	28295786,34	km
Slant range	R	28302163,62	km
Frequency	f	3,00	GHz
Path loss	Lpath	191,02	dB
Isotropic Signal Level at Ground Station		-173,72	dBW
Ground station			
Antenna Diameter	D	1,64	m
Antenna Efficiency	η	0,50	
LGA Gain	GRx	31,23	dBi
Ground station Losses between antenna and Rx	LRx	0,50	dB
Input Power at S-Band Receiver	PRx	-142,99	dBW
Downlink Temperature	T	34,21	K
		15,34	dBK
Figure of Merit	G/T	15,89	
Orbiter Signal-to-Noise Power Density	C/No	70,27	dBHz
System Desired Data Rate	R	3,00E+05	bps
		54,77	dBHz
Telemetry&Science System Eb/No for the Downlink	Eb/No	15,50	dB
Demodulation Method Selected	BPSK, QPSK		
System Allowed or Specified Bit-Error-Rate	10^-6		
Demodulation Losses	Ldem	1,00	dB
Eb/No Theoretical	Eb/No|theor	10,50	dB
System Link Margin		**5,00**	**dB**

Figure 10-5: Link Budget Case A: LEO to GEO link, S-Band

Table 10-3: Link budgets

Characteristic	Typical range
Antenna dish diameter	15m, 35m
Transmit frequency	
S-band	2025-2120 MHz
X-band	7145-7235 MHz
Receive frequency	
S-band	2200-2300 MHz
X-band	8400-8500 MHz
Telemetry (downlink)	
Normal data rate	up to 1 Mbps
Maximum data rate	up to 105 Mbps
Telecommand (up-link)	
Normal data rate	2 Kbps
Tracking	
Range accuracy	1 m
Range rate accuracy	0.1 mm/s

10.6. Re-Supply Items

- Communication equipment: lifetime 8-15 years

10.7. To Be Further Studied / Additional Considerations

- Study possible applications/payloads which could make use of high data rates provided by optical communications
- Study cost difference between baseline use of data relay systems vs. Ground stations
- Define internal communications of the station (e.g. LAN based on wireless technology, or cable/fiber optics)

10.8. Summary

At this early stage no specific requirements were provided to define any data transmission requisites, so the following assumptions were made:

- Maintain current ISS capabilities, as a minimum
- Channel communications via ground stations or data relay systems, and keep the other option as a backup

Based on the current ISS, three independent, purpose-dependant systems with exclusive frequency bands will be used as the baseline:

- S-Band: Command, Telemetry, Audio Channels
- K-Band: Payload, Video data, others
- UHF (2 independent systems): Extra Vehicular Activities, Docking

Two possible communication schemes where studied and provided as options for the customer:

- Communication channelled through TDRS/EDRS (inherited from ISS NASA concept, and being the preferred scheme), using Ground Stations as backup
- Communication channelled through Ground Stations, using TDRS/EDRS as backup

In addition, the: use of optical communications for high data rates was considered as an option which is available for both schemes (although at different data-rates).

11. Power

11.1. Requirements and Design Drivers

The stations Power Subsystem (PWR) shall be designed to handle the following requirements:

- Average power demand of 30 kW
- ISS-like orbit –maximum 36 minutes eclipse duration @ 400 km
- Survival Mode: 2 orbit completely without photovoltaic power
- Station lifetime: 15 years

11.2. Modes of Operation and Design Cases

The station will operate in the six different modes (see also Table 2-4):

- Default,
  - 25090.875W for 2 weeks,
- Standard electric,
  - 25090.875W for 2 weeks,
- Crew exchange
  - 27217.811W for 1 week,
- Docked
  - 25090.875W for 1 week,
- Survival
  - 12397.708W for 2 orbits
  - Standard Mode (normal operation before incident)
  - 36 minutes max. Eclipse → total loss of PV power for 1 orbit between 2 eclipses,
- Proximity operations
  - 25103.195W for 1 day.

The most energy demanding mode is the Survival mode which requires the battery to support the station for 2 orbits without any photovoltaic input. This mode will be used to size the battery.

The highest power demanding mode is the Crew exchange mode, but since the requirement for the station is 30000 W this value will be used instead to determine the volume of the solar panels.

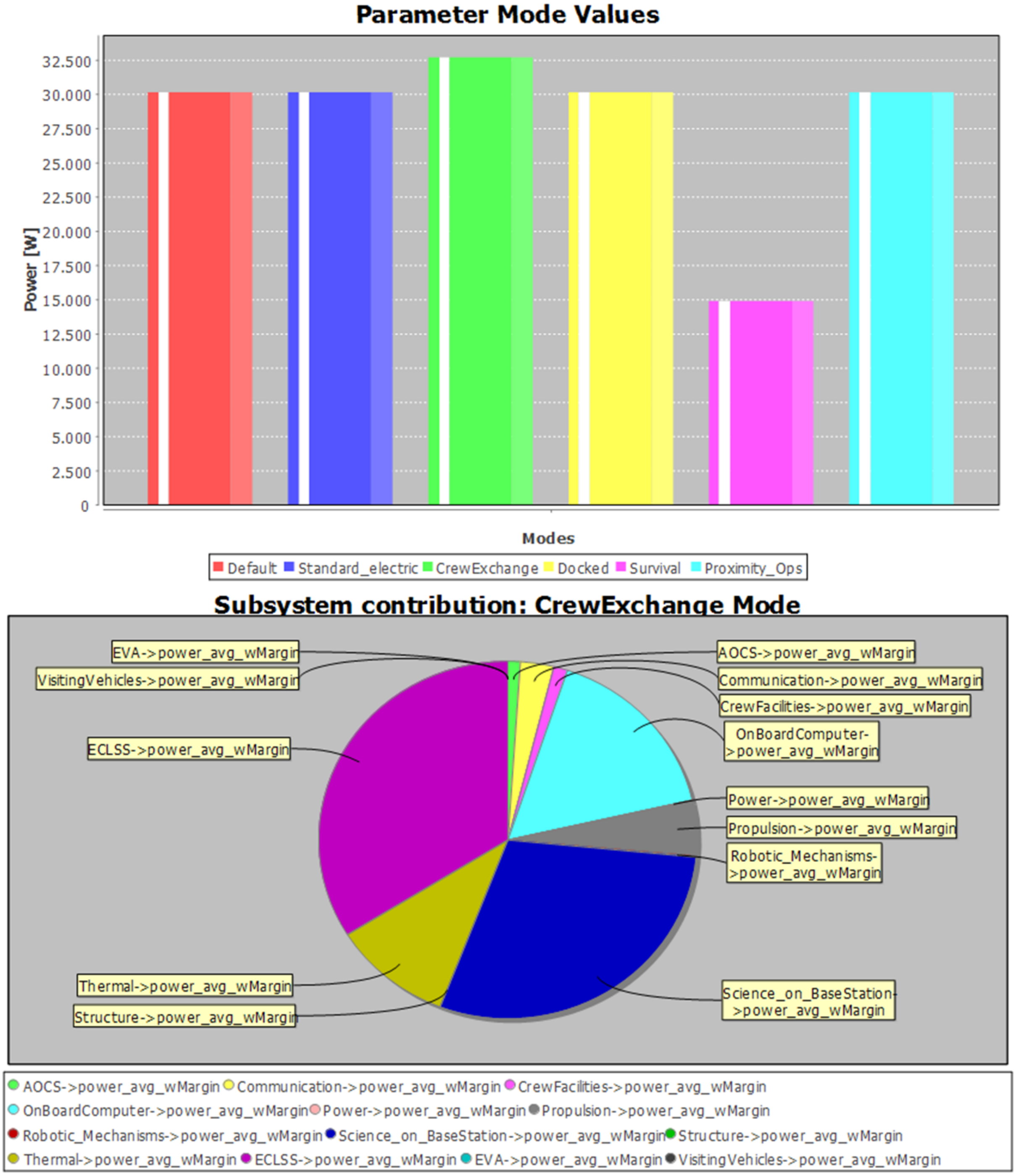


Figure 11-1: Overall power budget of the Base Station.

11.3. Power Budget

The following table (Table 11-1) shows the stations power requirements for the different subsystems during the six different operating modes. The values marked with orange are the powers required downstream from the battery, which is subject to photovoltaic efficiency, charging cycle efficiency, and power conversion efficiencies equal a system margin of 20%. This also includes Power Subsystem internal regulated power required for control equipment and communication with the on-board data handling. The values marked with red are the powers required by the Payloads and Subsystems.

Table 11-1: Overall power budget of the Base Station.

System Component	Parameter	Unit	Default	Standard_elect...	CrewExchange	Docked	Survival	Proximity_Ops
BaseStation	mode_duration	Second	1209600.000	1209600.000	604800.000	604800.000	345600.000	86400.000
▷ BaseStation **No margin**	power_avg_wMargin	Watt	25090.875	25090.875	27217.811	25090.875	12397.708	25103.195
◢ BaseStation **With margin**	power_avg_wMargin_wS...	Watt	30109.050	30109.050	32661.373	30109.050	14877.249	30123.834
◢ BaseStation	power_avg_wMargin	Watt	25090.875	25090.875	27217.811	25090.875	12397.708	25103.195
▷ AOCS	power_avg_wMargin	Watt	286.495	286.495	286.495	286.495	255.516	298.815
▷ Communication	power_avg_wMargin	Watt	781.200	781.200	781.200	781.200	748.800	781.200
▷ CrewFacilities	power_avg_wMargin	Watt	320.240	320.240	320.240	320.240	179.440	320.240
▷ OnBoardComputer	power_avg_wMargin	Watt	4442.400	4442.400	4442.400	4442.400	2810.400	4442.400
Power	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
▷ Propulsion	power_avg_wMargin	Watt	1364.780	1364.780	1364.780	1364.780	1364.780	1364.780
▷ Robotic_Mechanisms	power_avg_wMargin	Watt	27.000	27.000	27.000	27.000	0.000	27.000
▷ Science_on_BaseStation	power_avg_wMargin	Watt	7974.576	7974.576	7974.576	7974.576	761.520	7974.576
Structure	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
Thermal	power_avg_wMargin	Watt	2801.280	2801.280	2801.280	2801.280	2801.280	2801.280
▷ ECLSS	power_avg_wMargin	Watt	7092.904	7092.904	9219.840	7092.904	3475.972	7092.904
EVA	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
VisitingVehicles	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
BaseStation	margin_system	Percent	20.000	20.000	20.000	20.000	20.000	20.000
▷ BaseStation	power_energy_wMargin	Joule	30349922400.000	30349922400...	16461332092...	15174961200...	4284647815....	2168916048....
▷ BaseStation	power_energy_wMargin_...	Joule	36419906880.000	36419906880...	19753598511...	18209953440...	5141577378....	2602699257....

11.4. System Architecture

The baseline design for the Power Subsystem is based on the current ISS design but with increased efficiency and simplicity. The design of the power subsystem will assure redundancy using the following configurations:

- 3 times redundancy on the primary bus
- No redundancy, but double connections on the secondary bus
- DC/DC conversion done at P/L and S/S level to increase the system efficiency
- Massively parallel design on power generating system to increase the reliability.

The power sub system on the ISS uses a lot of steps for voltage down-conversion while this new design uses only two steps, one between photovoltaics and the battery for controlled charging, and one at payload or bus unit level. The voltage down-conversion is an essential part of the system since the relatively high battery bus voltage needs to be converted to the much lower voltages required by electronics and small actuators. Thus, the harness is operated at the highest voltage widely used in the power subsystem and its resistance contributes as little as possible to losses by voltage drop.

This new design is possible since the Li-ion battery cell chemistry proposed does not require any dedicated maintenance and conditioning operations, as the Ni-based battery cells originally used on the ISS. In fact, battery handling is virtually carefree as long as the minimum voltage, maximum voltage, and maximum current limits are observed. However, managing the average state of charge can significantly reduce battery ageing (reduction of maximum capacity) and thus increase useful battery lifetime [RD 4].

The design of the Power Subsystem can be seen in the figure above. This design consists of solar panels that are connected to a DC Switching Unit (DCSU) via Maximum Power Point Trackers (MPPT). The DCSU is a matrix of protected circuit breakers that connects different branches of photovoltaics, battery modules, and power buses. Also connected to the DCSU are the batteries and the Battery Charge/Discharge Units (BCDU) which in this design are not power converters but mainly switching units to manage (engage, disengage in a controlled manner) sub-units of the battery which is very large compared to other LEO spacecraft. (A byway power converter may be included to re-equalize the state of charge of a battery module for re-connection; however this is no operational power path.) Every module of the battery will be connected to its own BCDU to get more reliable system.

The power is routed by the DCSU and sent to the Main Bus Switching Unit (MBSU). The MBSU will distribute the power to all payloads and bus units that require it and of whom the majority will also do the second and final voltage conversion step themselves. For some power users requiring a common voltage at significant power level there may be centralized power conversion already in the MBSU and/or the MBSU may act as the controlled power switch output for other bus units towards a high-power actuator.

The MBSU can also be connected to other MBSUs, via the Transfer Converter (TC). The TC enables a link between the different MBSUs on the complete station, thus allowing other modules to easily dock and undock. An advantage of this concept is that only the amount of power that 'spills over' from section to section of the station has to undergo power conversion with a significant loss factor. If the docking/berthing interface of the station modules enables this, several power buses can run down the length of the station and into its branches, with each module's TC feeding to and/or drawing from any configuration of these buses through a set of selection switches. A representation of this can be seen in the figures below.

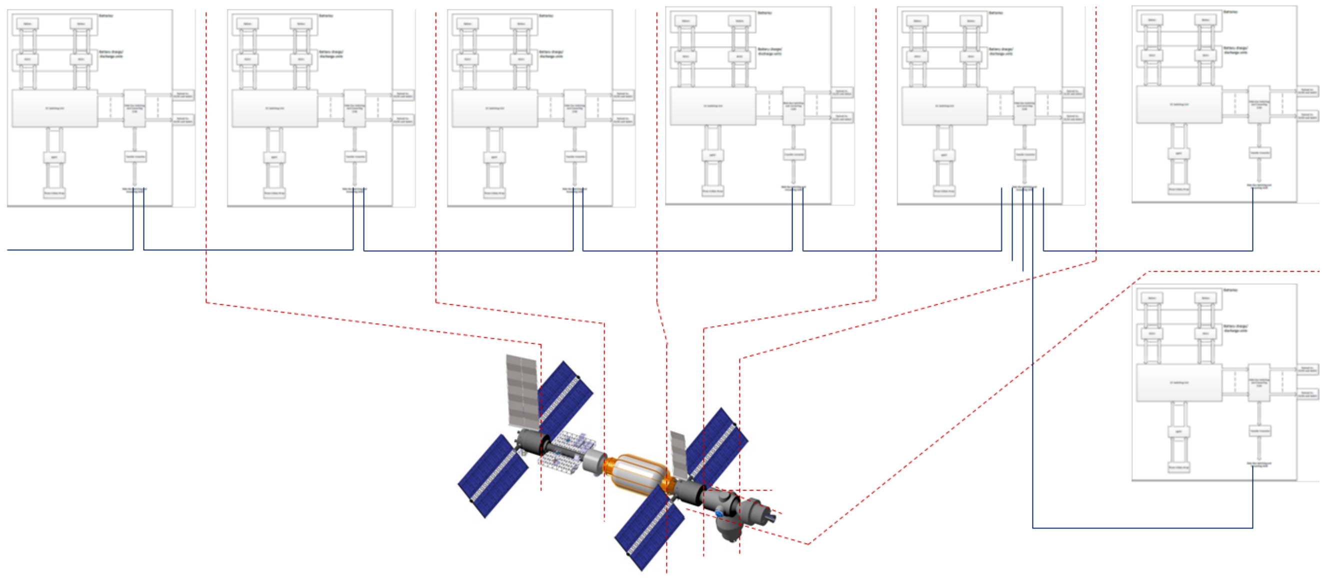


Figure 11-2: System Topology of the power subsystem with multiple modules.

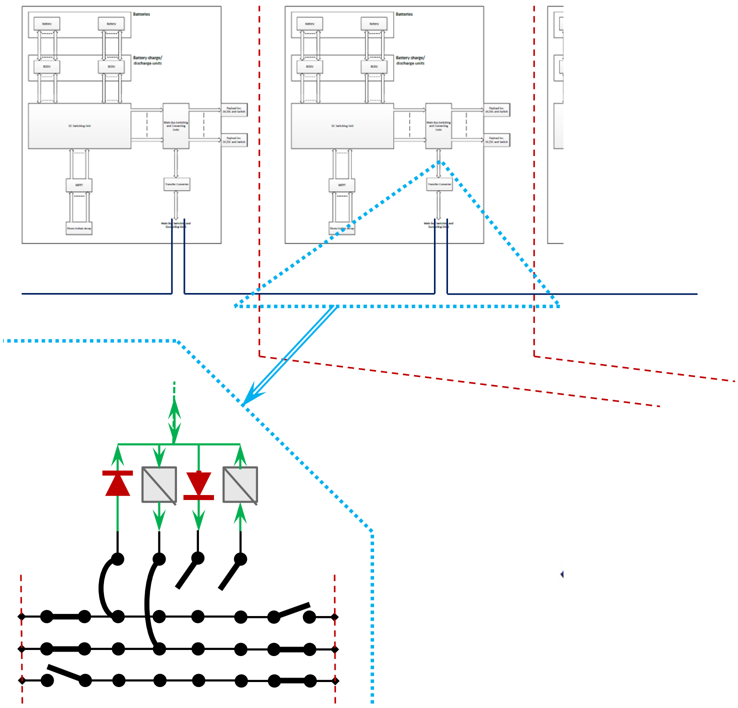


Figure 11-3: Detail of power transfer architecture concept

11.5. Power System Design

Some of the units used in the proposed system design are not available and have to be specially made for the stations unique requirements. The units that do not require this have been designed while the other ones have been scaled with the corresponding unit on the ISS.

11.5.1. Power System Losses

The losses in the power system have been estimated to the values seen in the figure below. The losses will vary depending on if the secondary power system is supplied directly from the solar panels during the sun phase or via the battery during eclipse. The different power losses in these two cases are:

$$Photovoltaic\ power\ losses\ ECLIPSE : P_{PV} \cdot 0.95^3 \cdot 0.8 = 0.6859 \cdot P_{PV}$$
$$Photovoltaic\ power\ losses\ SUN : P_{PV} \cdot 0.95 \cdot 0.8 = 0.76 \cdot P_{PV}$$

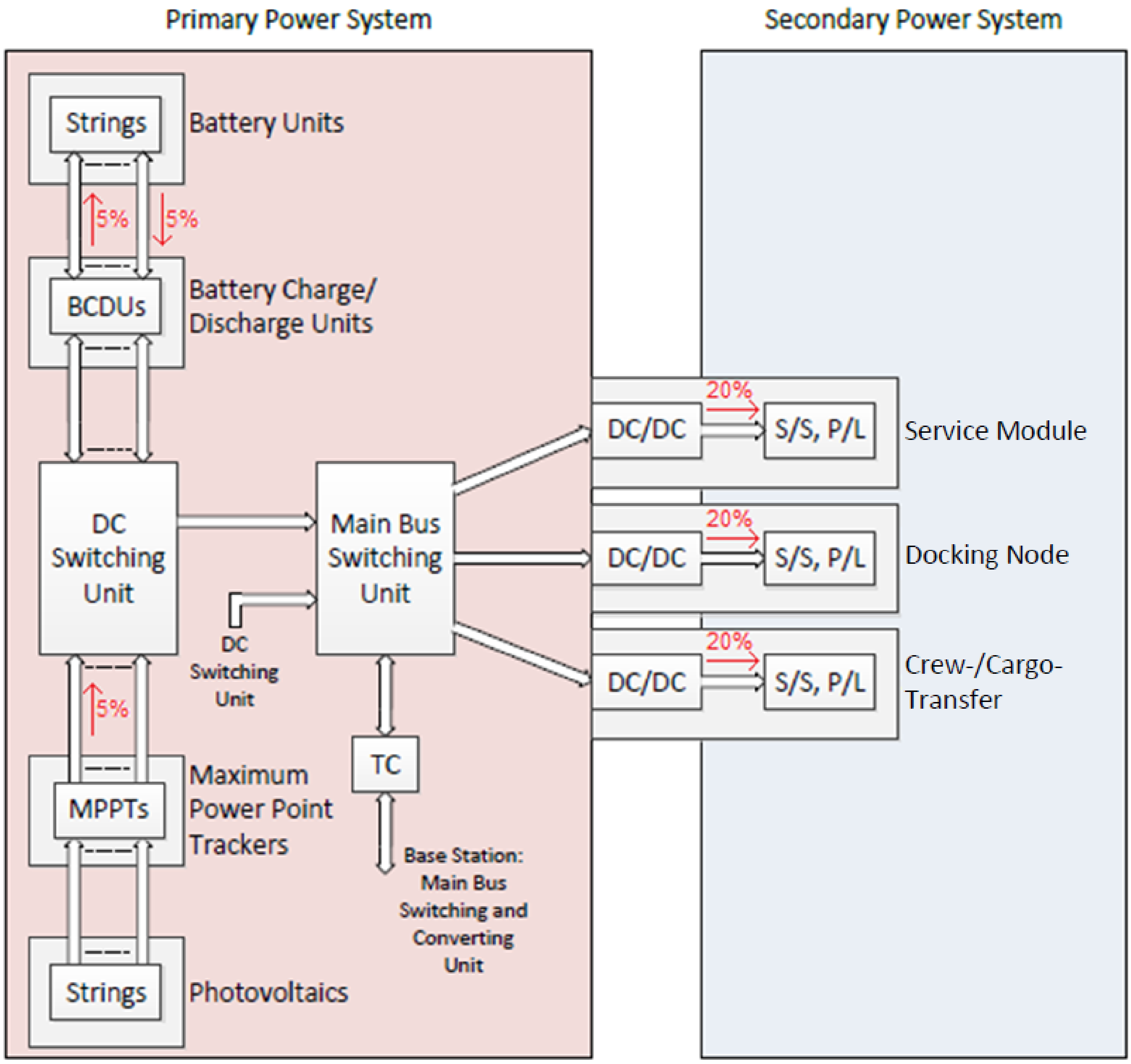


Figure 11-4: Power system architecture including losses.

The voltage at the DC/DC Converter Units (DDCU) inputs on ISS varies between 133-177 V, this minimum value was also selected for the Base stations power system. A minimum voltage of 133 V at the DDCUs would require a minimum Photovoltaic and battery output voltage of:

$$V_{PV}(min) = V_{Battery}(min) = \frac{133}{0.95} = \ 140\ [V].$$

11.5.2. Solar Panels

The selected solar cells are the Triple-Junction GaAs solar cells from AZUR SPACE. The cells have the following electrical characteristics:

- Cover glass and monolithic integrated bypass diode
- Worst case current (EOL 70°C): 0.49836 A
- Worst case voltage (EOL 70°C): 1.9436 V
- Mass per cell: 0.00356124 kg
- Area per cell: 0.003018 m²

Figure 11-5: Triple-Junction GaAs solar cell.

The amount of power that needs to be generated during sunlight to supply the power requirements during eclipse is calculated using the power requirement for the crew exchange mode, the power losses and the time duration for sun and eclipse:

$$P_{PV} = \left(\frac{32661.373 \cdot 55}{0.76} + \frac{32661.373 \cdot 36}{0.6859}\right) \cdot \frac{1}{55} = 68102\ [W]$$

The power requirement can then be used to size the total solar panel configuration in serial and parallel strings:

$$Cells_{Serial} = \frac{140}{1.9416} = 73\ cells$$

$$Cells_{Parallel} = \frac{68102}{73 \cdot 1.9416 \cdot 0.49686} = 968\ cells$$

The total amount of cells is then equal to:

$$Quantity_{PV\ cells} : 73 \cdot 968 = 70664\ pcs$$

Other parameters that need to be taken into account when estimating the total mass of the solar panels are specified in the table below. It is assumed that the Station will have two solar panels, each having a length of 30 m. The complete configuration for both solar panels will have a total mass of 690 kg. The Definitions as a percentage of the photovoltaic blanket mass has been scaled after the ISS. [RD 27]

Table 11-2: Solar panel mass estimation.

Unit Name	Mass [kg]	Definition
Photovoltaic blanket mass ($Mass_{PV}$)	251.6515	70664·0.00356124
Miscellaneous integration	172.3813	$Mass_{PV}$·0.685
Electrical equipment	172.3813	$Mass_{PV}$·0.685
Mast mass	93.3627	$Mass_{PV}$·0.371
Total Mass	**689.7768**	

The total area, using a cell spacing of 15%, is estimated to 250.9 m² according to:

$$Area_{PV} = \frac{70664 \cdot 0.003018}{0.85} = 250.8988m^2$$

11.5.3. Battery

The selected battery cells are the lithium-ion 18650HC cells from ABSL. The cells have the following electrical characteristics:

- Capacity of 130 Wh/kg
- Minimum cell voltage of 2.5V
- Operating temperature between -30°C → 60°C
- 2 cell-level safety devices; hard-short-safe up to 8s string, used by NASA JSC/GSFC on EAPU [RD 2][RD 3]

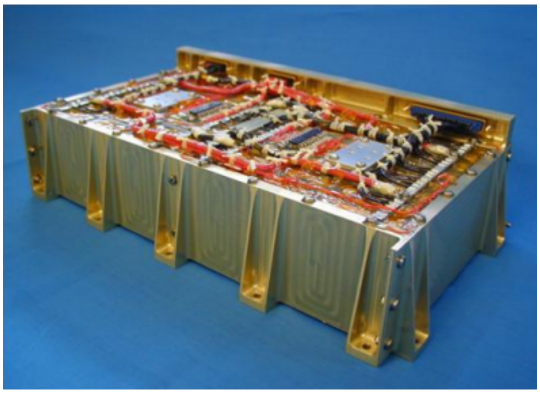


Figure 11-6: ABSL battery stack.

The battery is designed to have a lifetime of 15 years and throughout this time be able to deliver sufficient power during the two most power demanding modes:

- Requirement power including margin:

$$\frac{30000}{0.8 \cdot 0.95} = 39474W$$

- Survival mode including margin:

$$\frac{12397.708}{0.8 \cdot 0.95} = 16313W$$

A long mission lifetime puts though constraints on the battery that needs to be operated under optimal conditions to provide sufficient power at the end of the mission. This is achieved by selecting suitable end of charge voltage and depth of discharge values. For the selected cell different conditions have been tested by the supplier.

The battery will experience about 86600 cycles which will heavily decrease the capacity, while the huge amount of power requires a high depth of discharge to keep the battery as small as possible. According to these constraints the two most suitable cases from the diagram was chosen. These can be seen in the table below.

Table 11-3: Battery simulation cases, normal operation.

Case	End of Charge Voltage [V]	Depth of Discharge [%]	Temperature [°C]	Fade Rate [%]	End of Life Capacity [%]
1	4.05 (226.8 total)	40	20	1.4	29.5
2	3.95 (221.2 total)	30	30	1.3	32.3

The two cases were simulated using ABSLs battery simulation tool BEAST. Simulations were first made for the standard mode, confirming that the battery could reach a steady state for the charge and discharge level, also at the end of the mission. Then the safe mode was simulated to confirm that the battery would manage during a temporary solar panel pointing failure, and after that recover to the steady state. Since the Survival mode is the most demanding mode the depth of discharge for these simulations did not reach the maximum values defined above, but this will not change the results.

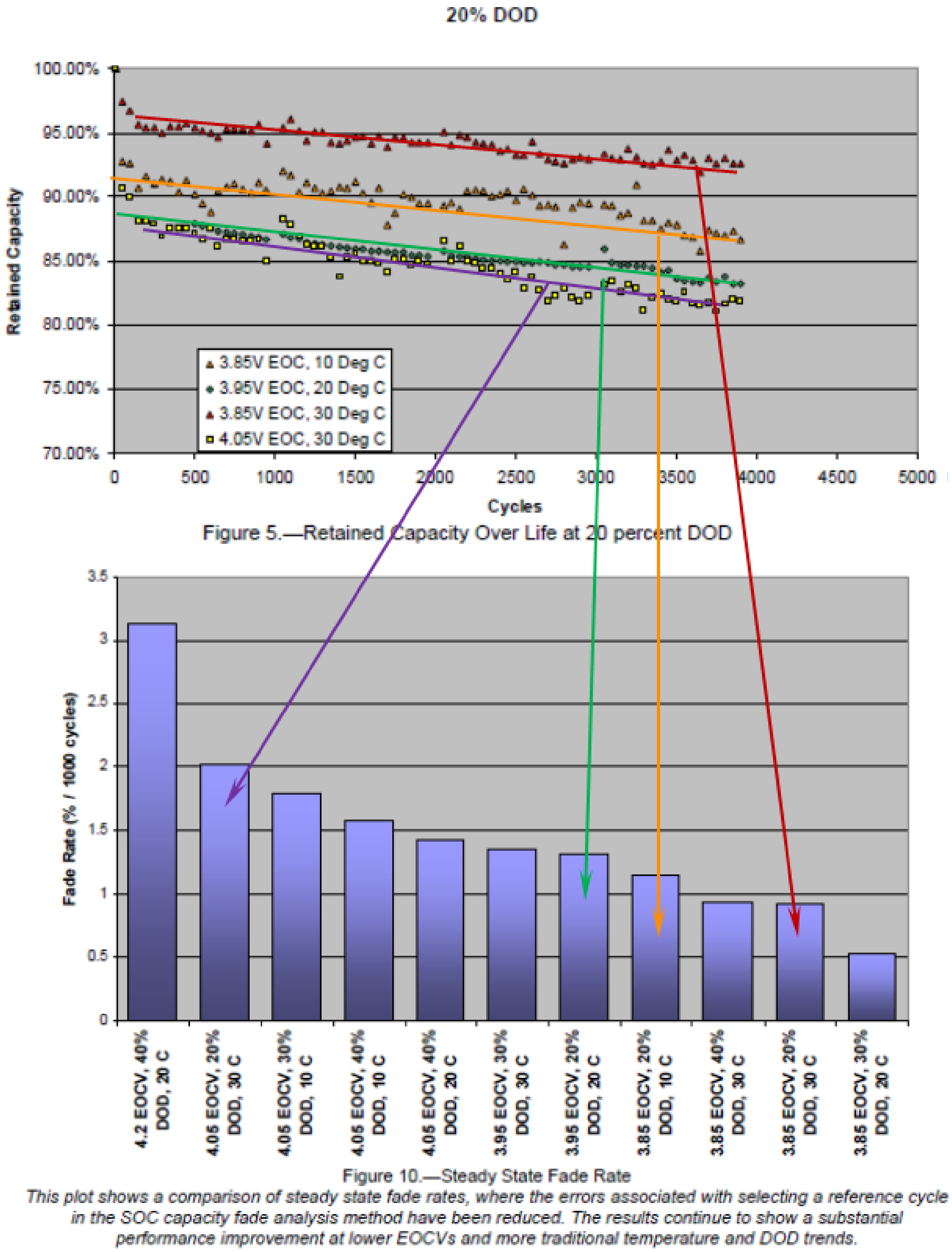


Figure 11-7: Retrievable capacity in the battery depending on discharge conditions [RD 4]

Case1 – Survival Mode:

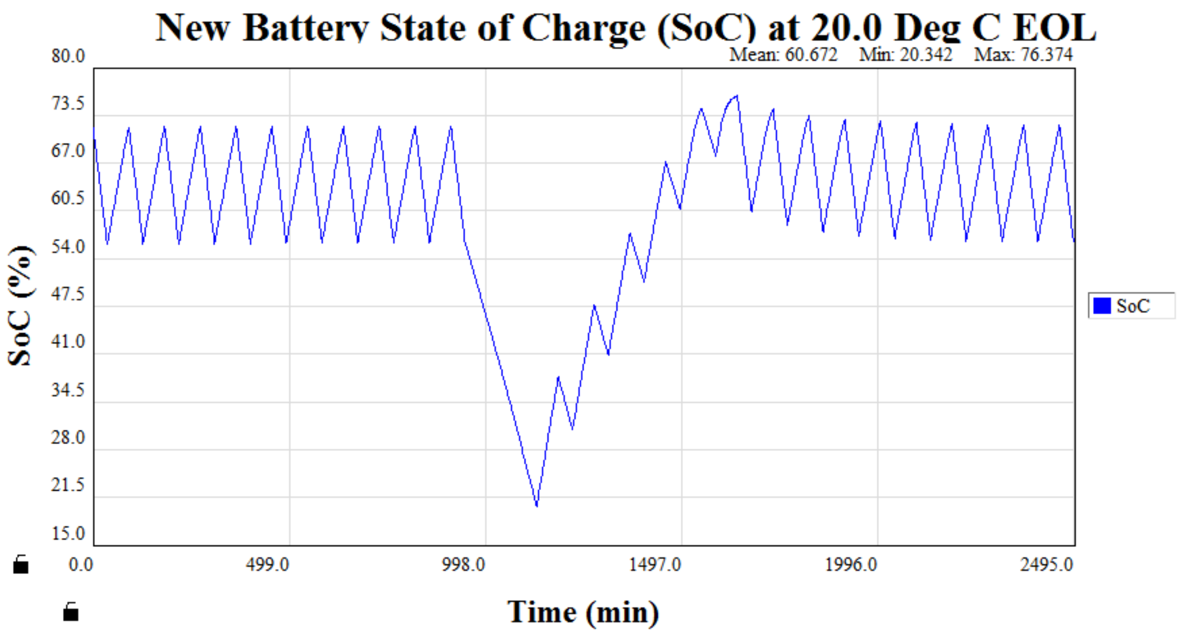


Figure 11-8: Battery SoC simulation – Survival Mode.

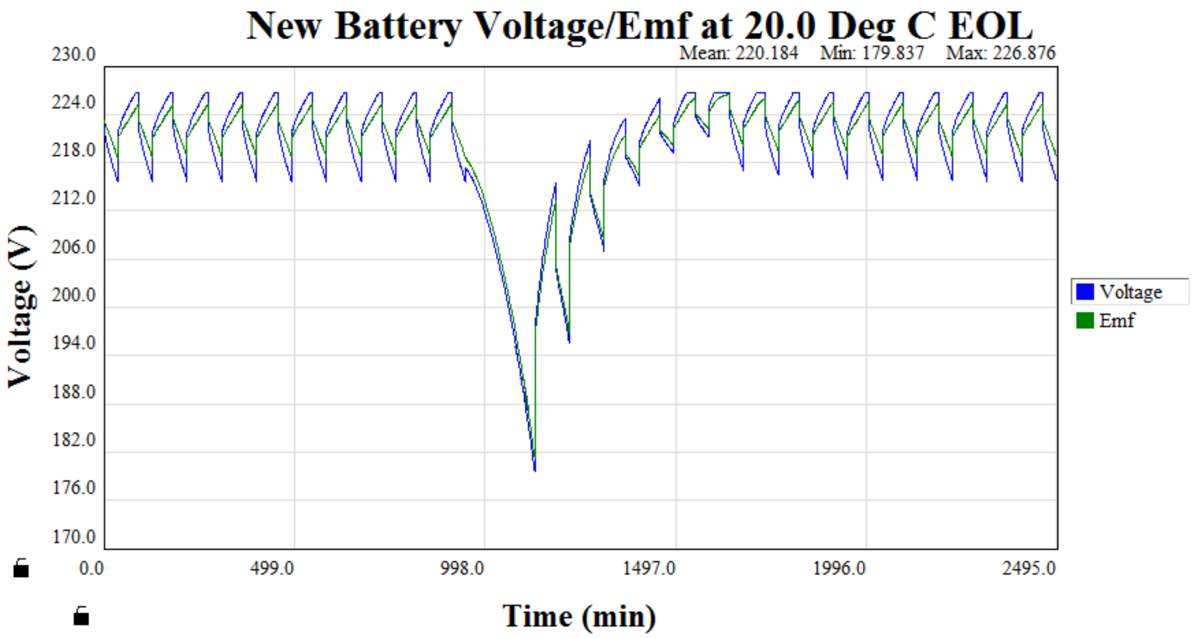


Figure 11-9: Battery voltage simulation – Survival Mode.

Case2 – Safe Mode

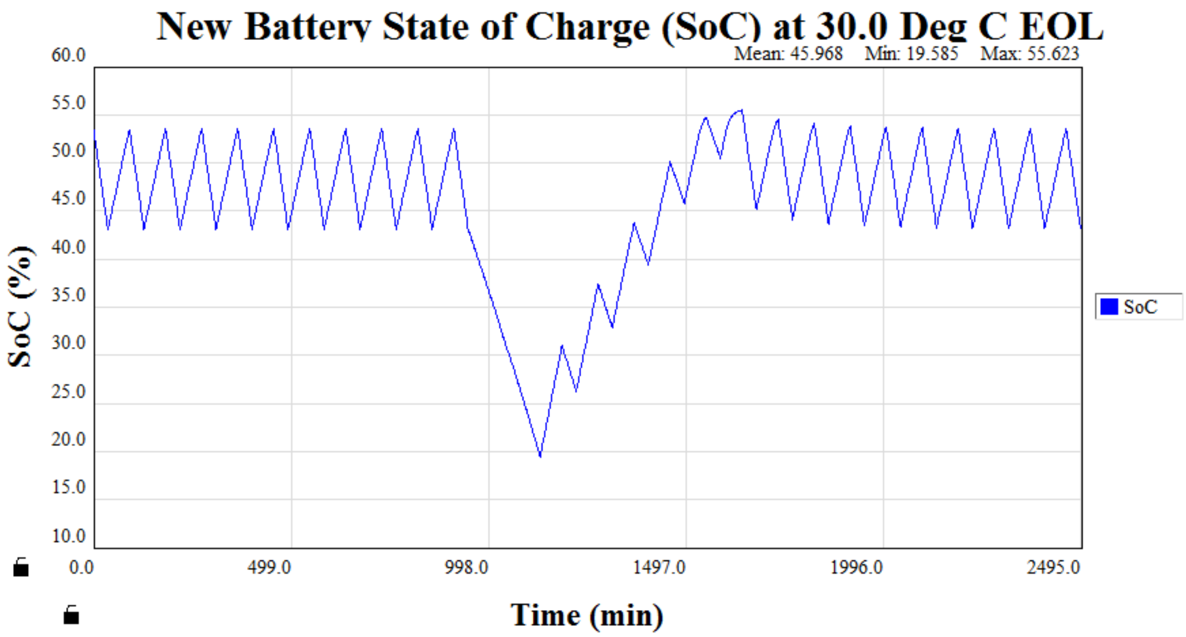


Figure 11-10: Battery SoC simulation – Safe Mode.

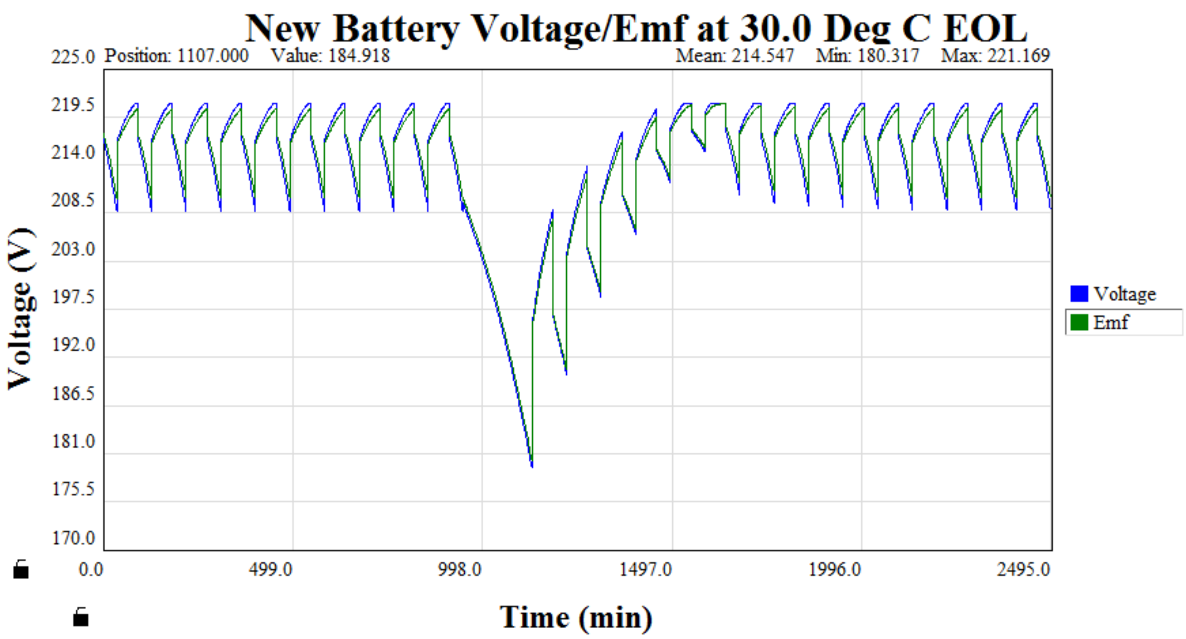


Figure 11-11: Battery voltage simulation – Safe Mode.

Simulation Results:

The simulations results can be seen in the table below. They clearly show that case 1 is the most suitable option since it has a much lower mass and still provides sufficient power. Case 1 is therefore chosen as the base line battery design.

Table 11-4: Battery simulation results – Survival Mode.

Case	18650HC Topology	Mass [kg]
1	56s1536p	4039
2	56s2200p	5785.04

Survival mode extension:

If the station should be manned it needs to have a longer survival-time than 2 orbits. The original requirement was a survival time of four days but this was not feasible since the battery becomes too large. The problem is mainly the high power demand, but also that the power needs to be supplied by only one battery assembly. If more modules where to be connected to the main station, with its own battery assembly, this would add a lot of lifetime to the main station.

In the table below are the results of simulations for battery sizes, depending on how much of the survival power one battery assembly needs to supply. If the station would increase in size by a factor of 4, the original requirement of four days would almost be meet. Compare the current battery size of 56s1536p with the battery needed to deliver a quarter of the survival mode power of 56s1694p.

Table 11-5: Survival Mode extension batteries.

Case	18650HC Topology	Mass [kg]
All power	56s6784p	17838.97
Half power	56s3392p	8919.49
Quarter power	56s1694p	4459.74

A possibility to increase the survival time before the battery grows could also be to take one of the batteries for the future modules to the station and use this as backup until the station grows. This could be possible if the power requirements were lowered.

Alternative Battery:

An alternative battery cell with the same mechanical shape and similar mass per cell could be the ABSL 18650NL, which has the capacity of 190 Wh/kg. Thus, the battery would become much more efficient per mass and volume than the selected cells, but these cells are relatively new and therefore only limited lifetime test information is available as it does not yet have the nearly 15 years of space operational history as the 'HC type.

Electronic box:

The electronic box will contain all the power equipment that will be needed on the Main station. All components needed in this box, except the battery, have been scaled to their equivalent unit on the ISS. The primary system, which includes everything except the DC/DC converters, will be placed in the service module while all three modules will have their own DC/DC converter enabling the power conversion at P/L and S/S level.

Table 11-6: Electronic box mass and volume estimation.

Unit	Number of units	Mass per unit	Volume per unit	Total mass	Total volume
Maximum Power Point Tracker (MPPT)	2	74.2	0.11	148.4	0.22
DC Switching Unit (DCSU)	2	63.7	0.13	127.4	0.26
Main Bus Switching Unit (MBSU)	1	117.8	0.26	117.8	0.26
Battery Charge/Discharge Units (BCDU)	2	245.7	0.51	491.4	1.01
DC/DC Converting Unit (DDCU)	3	33.9	0.05	101.6	0.16
Battery	24	168.3	0.17	4039	4.08
Total, Primary system (Secondary system)				**4924 (+101.6)**	**5.83 (+0.16)**

The total mass for the primary system including margin is estimated to:

$$Total\ mass\ with\ 10\%\ margin\colon\ M = \frac{4924}{0.9} = 5471kg$$
$$Total\ volume\ with\ 20\%\ margin\colon\ V = \frac{5.83}{0.8} = 7.29m^3.$$

The mass and volume for the secondary system is estimated to 102 kg and 0.16 m³ respectively.

11.6. Options and Trades

11.6.1. System Topology (massively) parallel design

The primary system design will be focused on a massively parallel topology, which will provide a very stable and fault tolerant system. This design is illustrated in the figure below. The design will be focused on the following aspects:

- panel level – 4 … 16 blocks corresponding to mechanical panel structure segments
  - largely traditional design with modular redundancy
- kW / kWh level – 10's of parallel blocks with ~1 kW photovoltaic power
- single PV string – 100's of parallel blocks with ~50 W photovoltaic power
  - integrate DCSU switch in each MPPT-BCR (no separate DCSU)
  - graceful degradation

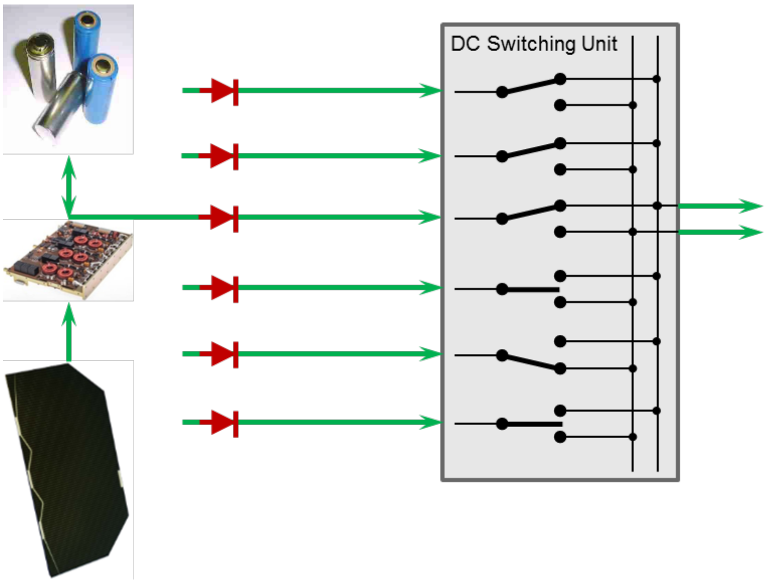


Figure 11-12: System Topology option of the power subsystem (parallel design).

11.6.2. Photovoltaics: rigid vs. thin-film

Lightweight photovoltaics structures had their part in the ISS history: the ISS panels themselves are a "semi-rigid" structure, rigid photovoltaic cells mounted on a sectioned foldable substrate. Also, the Hubble Space Telescope (HST) has used "semi-rigid" panels with the following data:

- 2 * 2.2 kW with 14% Si cells,
- Molybdenum interconects and silicone-coated Kapton base film,
- 12.2 x 2.5 m,

- -100°C…+100°C, 30kcycles, 5 years
    - originally: 2 * 2 kW, 12.7% Si cells, silver-mesh strip interconnects, fibreglass-reinforced Kapton – neither silver nor uncoated Kapton is AtOx resistant → design change
    - after retrieval: minimal power degradation ("could have been relaunched")
- HST new rigid panels: 1/3 less area, 20% more power – derived from Iridium spare panels, 2 * 2.8 kW

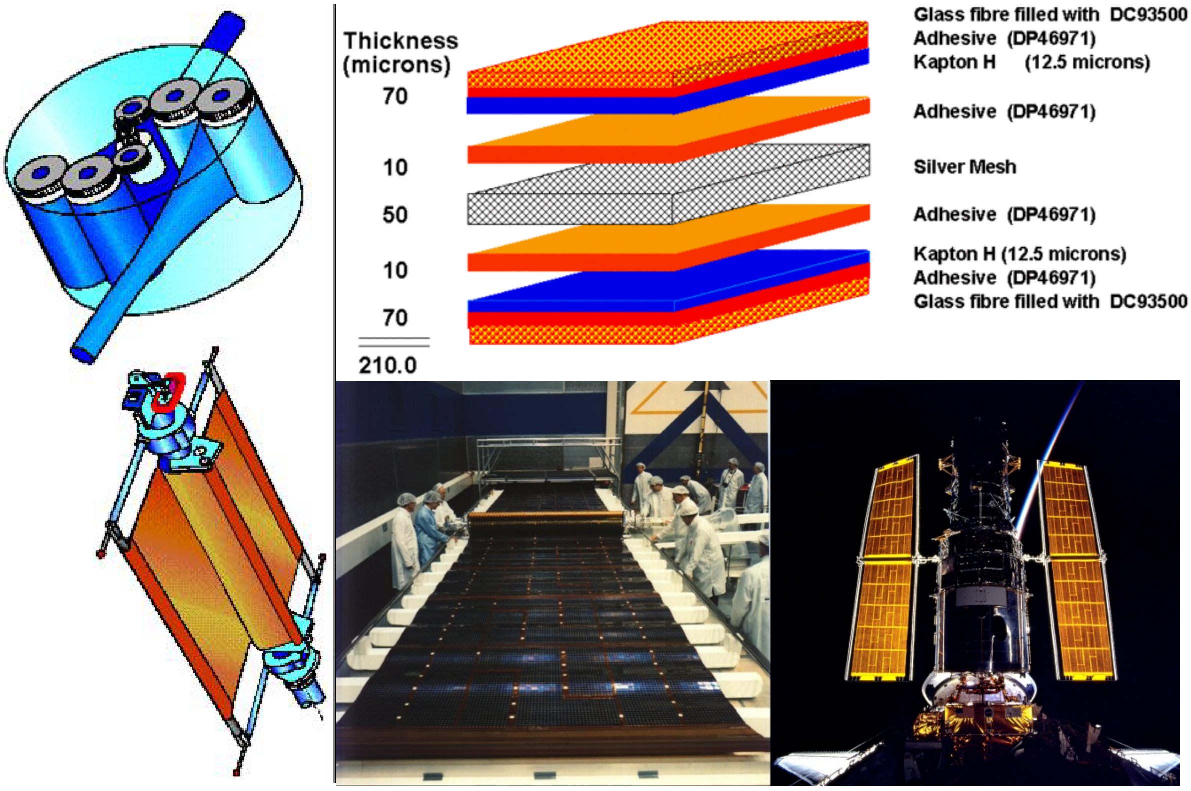


Figure 11-13: Hubble Space Telescope semi-rigid photovoltaics; booms and panel base foil.

In the course of the DLR-ESTEC GOSSAMER Roadmap to Solar Sailing, extremely lightweight fully flexible photovoltaics have been studied, first as a method to supply moderately sized pure sailcraft at power levels of several 100 W, typical of interplanetary probes. Renewed interest in high-power 'solar power sail' missions [RD 5][RD 6] and the success of the first interplanetary solar sail and flexible photovoltaics demonstrator, IKAROS, [RD 7][RD 8] and the rising power demand of geostationary satellites has refocused

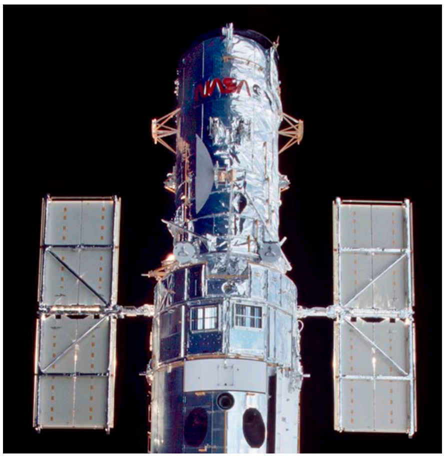


Figure 11-14: Hubble Space Telescope on rigid photovoltaics.

the GOSSAMER developments towards structural and electrical design for very large photovoltaic arrays, using all the elements pioneered in the original GOSSAMER-1 sail deployment demonstrator design.

The baseline photovoltaic cell used in GOSSAMER is the Copper-Indium-Gallium-Selenide (CIGS) cell commercially produced by Solarion AG, Leipzig/Dresden. Below, this large-area cell is compared with the state of the art of Si cells, as used on ISS and HST, and of triple-junction cells as used on almost all modern spacecraft:

Table 11-7: Photovoltaic cells comparison

semiconductor stack	CIGS	Si	triple junction
cell type	Solarion standard	Azur S32	Azur 3G30A with coverglass
efficiency, %	11.5	16.9	29.3
area/cell, cm²	55.25	23.61	30.18
standardized measurement conditions	**1000** W/m² 25°C AM1.5G	1353 W/m² 28°C AM0	1367 W/m² 28°C AM0
$U_{MPP(STC)}$, V	0.389	0.528	2.409
$I_{MPP(STC)}$, A	1.643	1.025	0.503
$P_{MPP(STC)}$, W	0.639	0.541	1.211
$U_{OC(STC)}$, V	0.536	0.628	2.690
$I_{SC(STC)}$, A	1.903	1.081	0.520

As can be seen, CIGS cells operate at somewhat lower voltge and higher current per cell, but deliver power of the same order of magnitude. Being a newer development, the efficiency of CIGS cells still lags behind other single-juction cells but is eventually expected to catch up. Unlike other photovoltaic cells, CIGS cells appear nearly immune to radiation-induced degradation as demonstrated by the Japanese MDS-1 mission [RD 9] since 2002 crossing the radiation belts on a geostationary transfer orbit (GTO) [RD 10] on a components dosimetry and radiation characterization mission [RD 11].

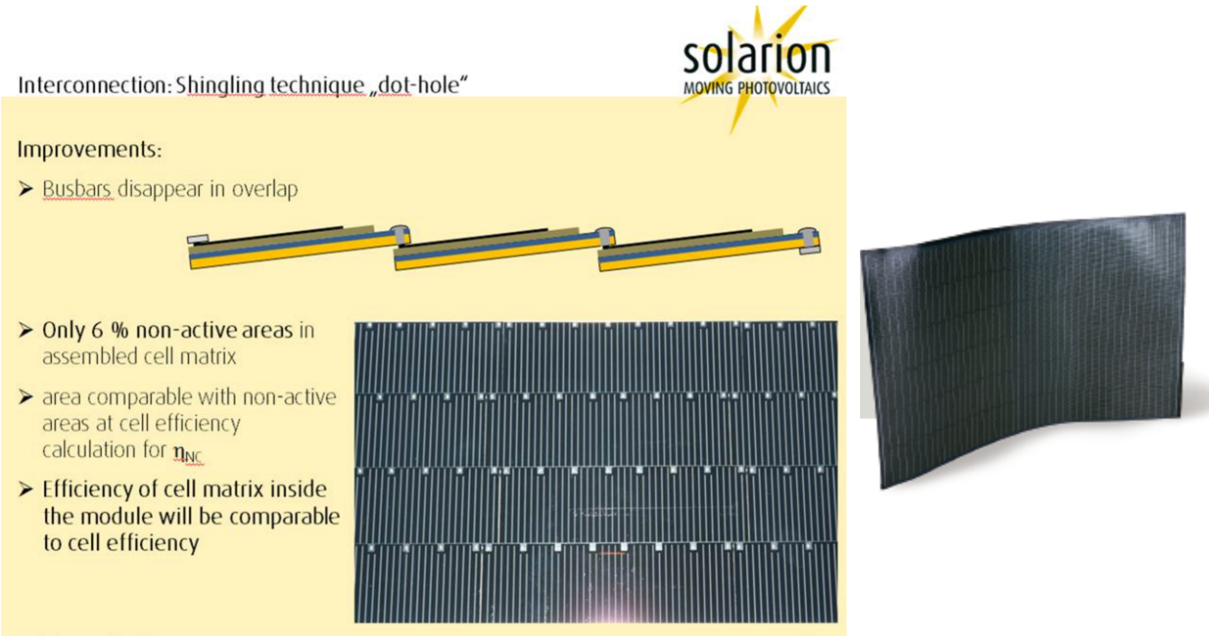


Figure 11-15: CIGS cells: Solarion company mounting method (left) and commercial flexible panel (right).

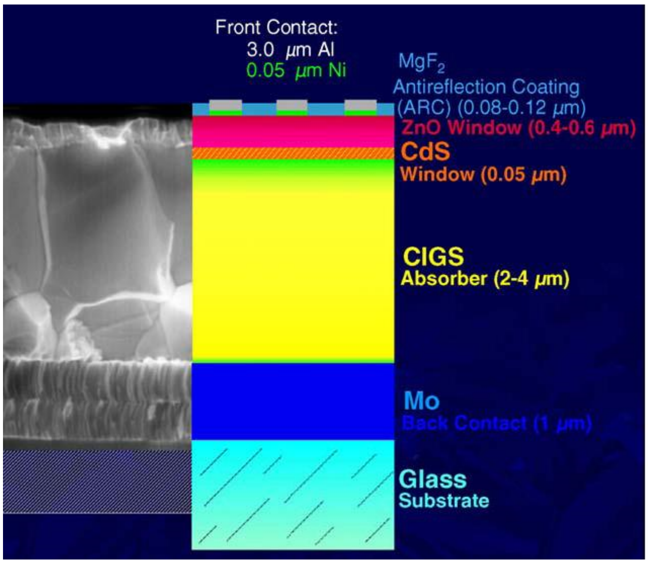


Figure 11-16: CIGS cells: typical CIGS semiconductor stack (via wikipedia CIGS entry) – note that here the glass substrate is replaced by a flexible polyimide foil.

The voltage levels chosen for this study suggest the following photovoltaics panel topology for a CIGS-based solution:

- 100 V with 0.290 V @ 95°C → 344 cells/string → 9.99 m/string
- ~31 V with 0.290 V @ 95°C → 107 cells/string → 3.11 m/string
  → rule of thumb: approximately 10 cm/V
- string width ~0.193…0.195 m
- coverage >90% active area
  - triple-junction: ~2.1 cm/V , width ~0.081 m

Due to the higher number of cells required and the relatively large-area standard CIGS cell, this approach seems less flexible in the utilization of small areas or the division of a given panel area. However, CIGS cells can be cut to size (*sic!*) producing narrower strings at the same voltage-defined length, and they can also be cross-shingled to generate a homogeneously filled photovoltaic area, effectively one wide string of voltage-defined length (see photograph in the shingling technique figure above).

The point of departure performance expected based on the existing GOSSAMER-1 hardware for the photovoltaic-oriented follow-on GOSOLAR proposal are as follows:

- photovoltaic: 500 g/m² – CIGS cells, conductors, base foil
  - in solar sail application: 11 g/m² – Al on 7.5 µm Kapton foil
- boom mass: ≥1…2 kg/m depending on docking loads
  - <u>TBC for heavier-than-sail PV foil,</u>
    - ISS booms: 4…5 kg/m

Key to the mass efficiency gain expected from the GOSSAMER-based technology is the two-dimensional deployment of a square thin film structure characteristical of solar sails. The confidence limit on the GOSSAMER deployment technology studied so far for solar sails is at quadrants of a 50…70 m square structure.
The transition from mainly thin-film sail to fully photovoltaics-covered foil needs further study which is the subject of a proposed project intended for the year 2016 and beyond. However, the original GOSSAMER Roadmap is currently being wrapped up by the construction of a QM-level Boom Sail Deployment Unit (BSDU) and a ground-based deployment qualification campaign till the end of 2015, proving the controlled balanced deployment concept which is key for all large lightweight boom-supported structures.

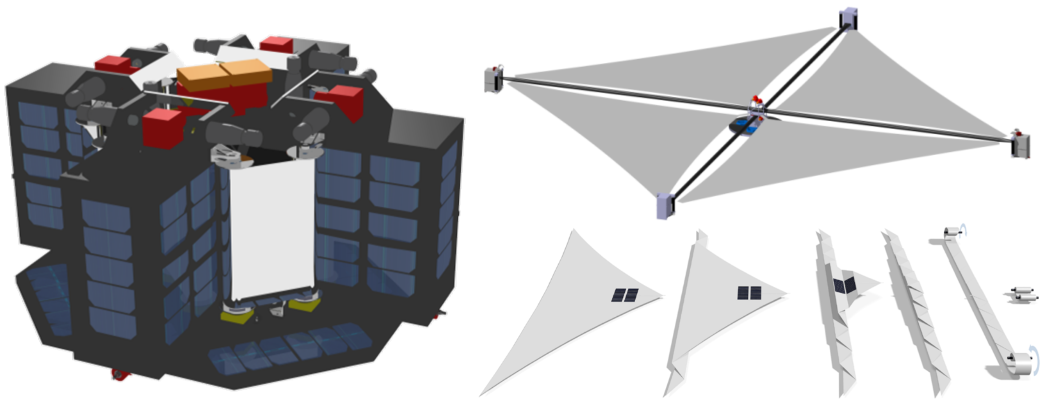


Figure 11-17: The GOSSAMER-1 solar sail deployment demonstrator spacecraft design status at the mid-2014 point of project reorientation and the folöding sequence of a quadrant.

The large lightweight structures deployment technology of GOSSAMER can be adapted to the conventional solar panel on a boom concept using one or two quadrants mounted at the central boom node, or use the four-quadrant sail-like structure mounted at one tip. The mounting point can be atop a boom or, in the case of the 1- and 4-quadrant concepts, on the spacecraft structure as the tip angle enables 45° clearance from the mounting point. However, only the 4-quadrant design requires no additional rigging to balance the bending moment on the booms as required for lightweight design. In the other variants, the compensation rigging has to lead to the solar panel turntable.

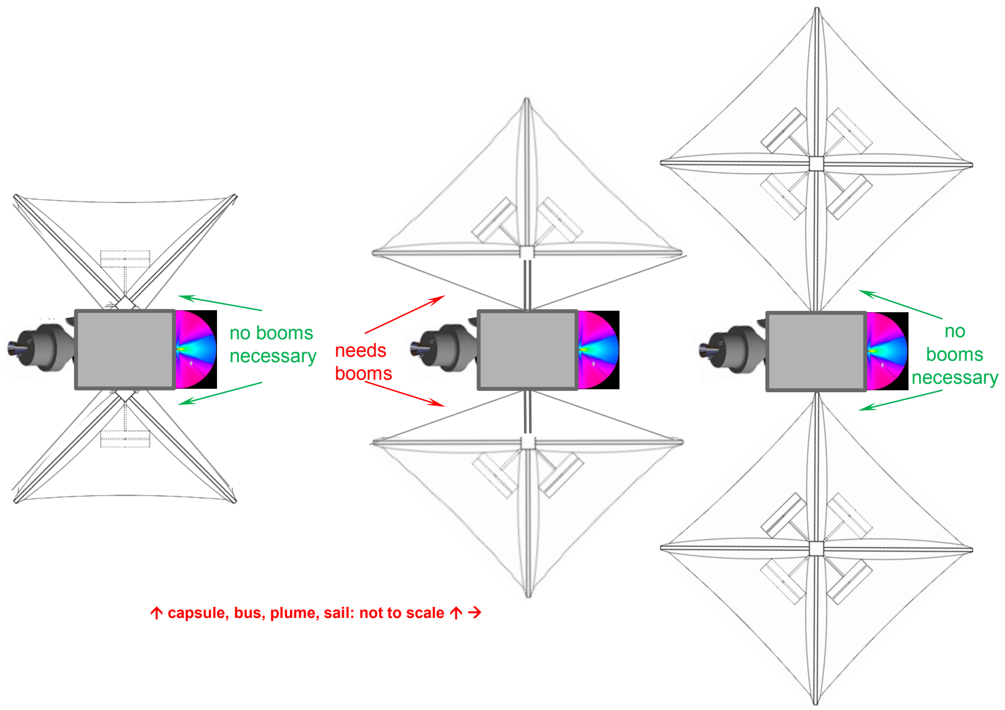


Figure 11-18: GOSSAMER-type solar panels concepts: 1-, 2-, and 4-quadrant.

A first iteration leads to a two 4-quadrant square lightweight solar panels design with the following parameters:

- approximate size based on <u>*immediately available*</u> (i.e., on stock for Gossamer-1) 10.2% efficient CIGS cells :
  - two 15 m square (or one 22 m square) „PVsails“ for 54 kW
    → well within the 50…70 m-square confidence limit for sail structures
- *ad-hoc* integration concept
  - one Boom Sail Deployment Unit (BSDU) per panel attached at bus, on PV turntable
  - 3 BSDUs per panel can be discarded (if compatible with space debris Code of Conduct)
  - end clearance given by 45° tip angle – no long stand-off boom as for rectangular PV panels

Note that a detailled design was not yet studied in the Gossamer, HPA or upcoming GoSolAr projects.

Summary & comparison:

- baseline assumption for rigid panels: 185 m², ~920 kg, ~54 kW –~17 kg/kW → 58 W/kg
- large lightweight deployables estimate: 36 m² (6x6) array based on Gossamer-1 – 5 kg/kW → 200 W/kg

- photovoltaic foil: 500 g/m² – CIGS cells, conductors, base foil – 150 μm (200…300 μm on joints)
- boom mass: 1.5 kg/m TBC for PV foil equipped structure, depending on docking loads
- CIGS cells available: 10.2% efficiency in standard conditions ≈ 3 times area required compared to TJ PV
  → maybe necessary to trade with drag-related propellant mass included

11.7. Mass and Power Budget

11.7.1. List of Equipment

Table 11-8: Mass budget of the power equipment.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
HA_Battery	168.63	20.00	33.73	202.36	4.69
HA_PowerSystemComponents	1000.00	20.00	200.00	1200.00	27.83
HA_Solar_panels	48.00	20.00	9.60	57.60	1.34
Harness	65.22	20.00	13.04	78.26	1.81
SM_Battery	1027.95	20.00	205.59	1233.54	28.60
SM_PowerSystemComponents	1000.00	20.00	200.00	1200.00	27.83
SM_Solar_panels	284.00	20.00	56.80	340.80	7.90

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	3593.80			4312.56	

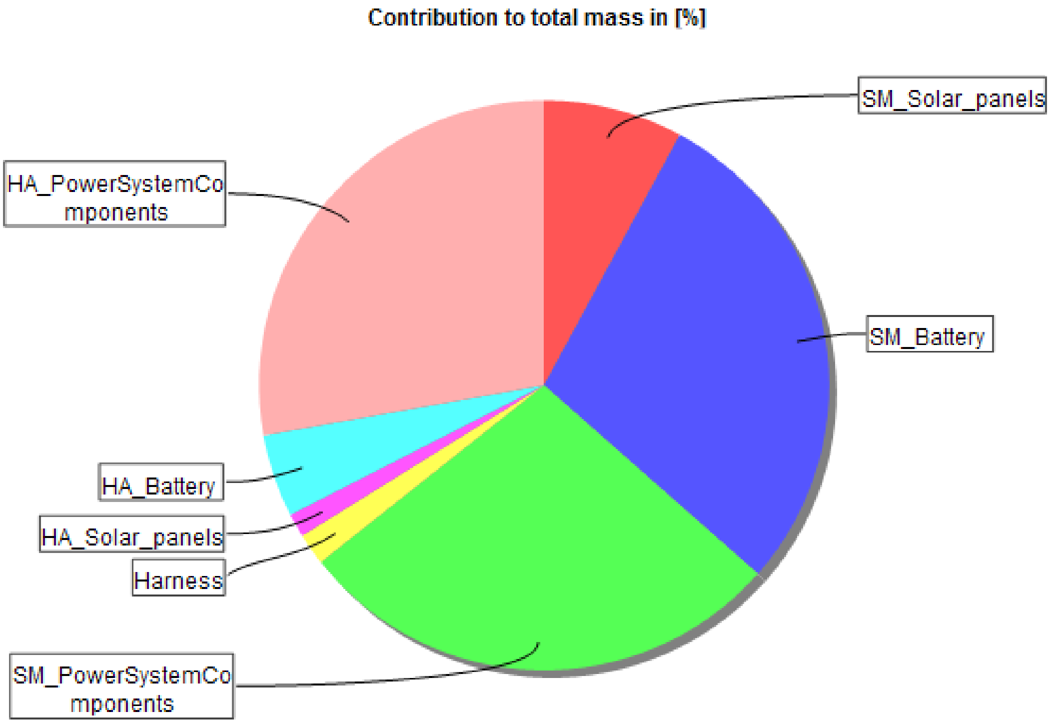


Figure 11-19: Pie chart of mass distribution of the Power subsystem.

11.7.2. Power Budget

The power consumption of the Power subsystem has been covered by the power system layout.

11.8. Re-Supply Items

- none (baseline with REoCV regime on battery)
  - maybe 1 battery exchange after 7.5…10 years (if no REoCV)
  - alternative: increase battery somewhat more ⇔ reduce DoD

11.8.1. Battery Ageing

Ageing data on ABSL18650HC are based on real measurements:

- ~80000 cycles observed by now (plots here ~mid-2007)
- significant extension can be expected from Reduced End-of-Charge Voltage (REoCV) regime
- possible to have battery with ≥15 years lifetime
  - +20…+30°C
  - <<20% Depth of Discharge
  - REoCV applied

11.9. To Be Further Studied / Additional Considerations

The proposed design needs to be further studied, especially focusing on the further developments of the ISS design. The major things that need to be studied are:

- The new configuration using less converters in the power flow than on ISS
- The cross-feeding between modules, topology optimization for various docking configurations of station modules & possible growth
- Massively parallel PV – MPPT – Battery design (partially assumed)
- A more accurate estimation of the mass and volume for the Electronics box.

11.10. Summary

The power requirements for the Post-ISS Study have been fulfilled. This was accomplished using an ISS inspired Power system, but modifying it to a simpler but still as safe and robust system. The new design will also make it easy for the station to develop and grow over the years.

One major design change in the design was the battery. A lot of progress has been made in this area since the ISS was built and todays batteries are much more stable and do not require as much power control for stabile operations. The removal of the many voltage conversion steps will save a lot of power that otherwise would have been lost as heat.

Another improvement was the solar panels that with today's technology have a higher efficiency and does not require the same huge areas as before.

The units required in addition to the battery and solar panels could probably also be smaller than on the ISS, shrinking the mass and volume of the electronic box further, and should be investigated in the future.

12. Thermal

12.1. Requirements and Design Drivers

The Thermal Control System design shall meet the thermal mission requirements, the thermal performance requirements and the interface requirements to other subsystems. The design drivers for the TCS are listed below based on [RD 12]:

- Mission Requirements
  See 2.1 Mission Requirements
- Definition of Orbits
  See 8 Mission Analysis
- Timeline and Load Cases
  See 11.2 Modes of Operation and Design Cases
- Spacecraft Geometry and Coordinate System
  See 3 Configuration
- Temperature Limit, Gradients and Stability Requirements
  Because of the maturity of current study, component level temperature limit, gradient, and stability requirements are not defined yet. However, all the components which are located inside the space station shall be balanced within the manned mission temperature level.
- Properties of Spacecraft Equipment
  Because of the maturity of current study, component level thermal properties are not defined yet. Thermal analysis considering the thermal properties of each component shall be performed in the future study.

12.2. Baseline Design

This clause describes the thermal control concept based on the Thermal Control System requirements. Because of the large heat generation and temperature range requirement for a manned mission, an Active Thermal Control System is required for a space station Thermal Control System design. An Active Thermal Control System uses a mechanically pumped fluid in closed-loop circuits to perform three functions: heat collection, heat transportation, and heat rejection. The Active Thermal Control System consists of External Active Thermal Control System and Internal Active Thermal Control System.

12.2.1. External Active Thermal Control System

External Active Thermal Control System is the primary active heat rejection system on the space station. It collects, transports, and rejects excess heat from all space station modules. The External Active Thermal Control System contains complete two ammonia coolant loops and each space station module has a heat exchanger or a cold plate from both coolant loops for redundancy. The system diagram is shown in Figure 12-1. [RD 13]

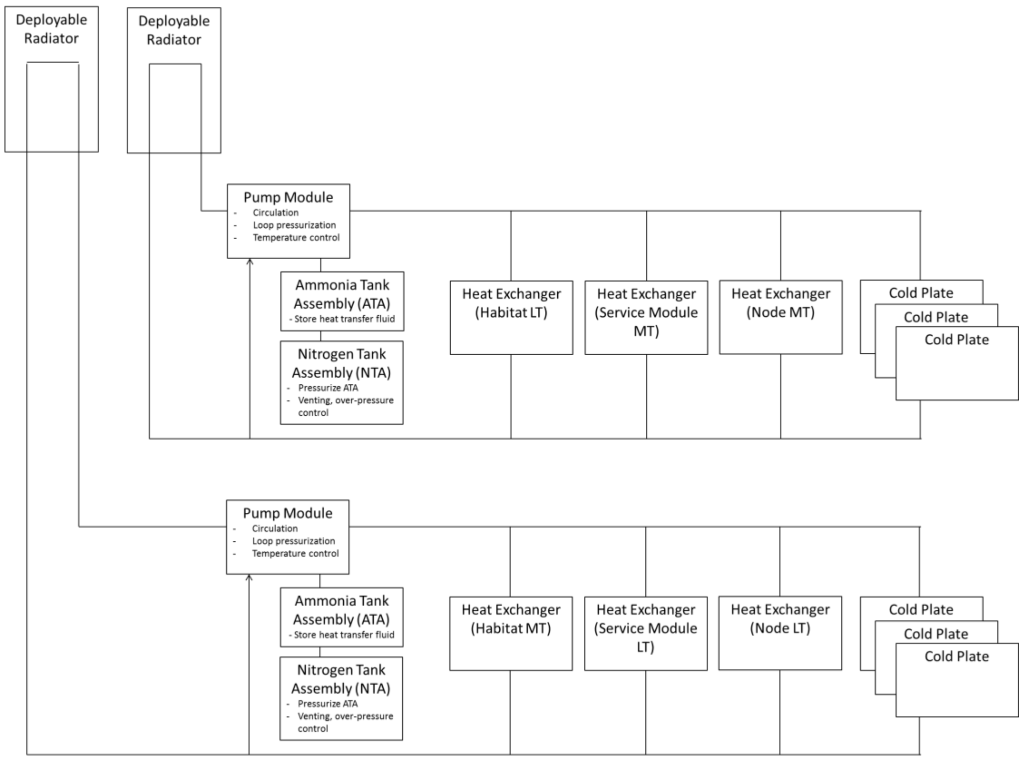


Figure 12-1: Layout of the external fluid loop of the thermal subsystem.

12.2.2. Internal Active Thermal Control System

Internal Active Thermal Control System consists of loops that circulate coolant fluid through the interior of each module to collect the excess heat from equipment and transfer this heat to External Active Thermal Control System via Interface Heat Exchangers. For the coolant fluid type, water is considered as baseline, because it is an efficient thermal transport fluid, is safe inside a habitable module, and has design, development and operation experiences from ISS. In each space station module, separate Internal Active Thermal Control System is to be accommodated. With regard to the Internal Active Thermal Control System of ISS, a coolant loop has redundancy, but the design concept is different depending on a module. Figure 12-2 shows two design options used in ISS modules [RD 14]

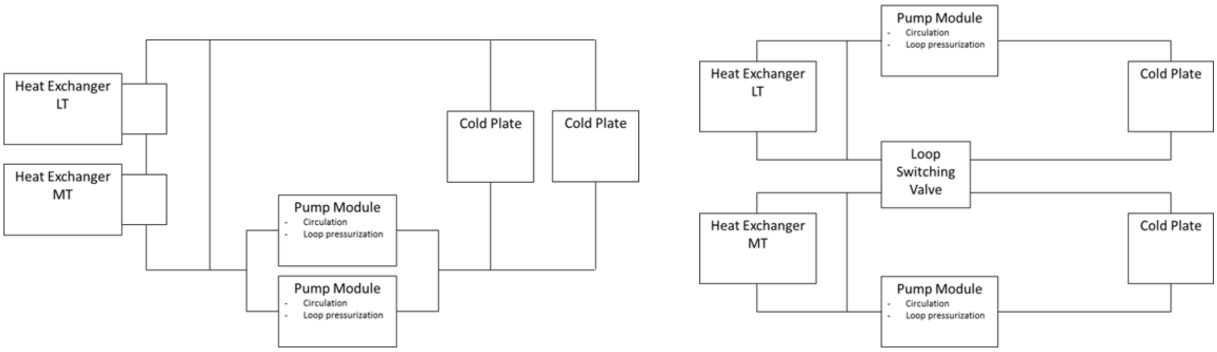


Figure 12-2: Redundancy concept of the internal fluid loop (left: Columbus based; right: JEM based)

12.2.3. Radiator

Heat collected by the External Active Thermal Control System is rejected to space by radiators. In order to realize large surface area and efficient heat rejection capability, deployment mechanism and rotation mechanism are considered as baseline design. For preliminary radiator feasibility study, the following parameters/conditions were used and the calculation result is summarized in Table 12-1:

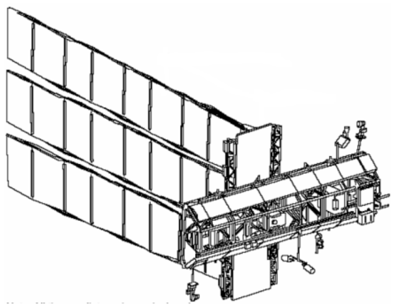


Figure 12-3: ISS thermal radiator [RD 13]

- Radiator with rotation capability (which prevents direct sun input to the radiator)
- Surface Coating: White paint with α=0.15, ε=0.9 [RD 15]
- Radiator Size: 90 m^2 (2 x 45m^2)

Table 12-1: Radiator design.

	Heat Input [W]	Environmental Input	Stability Temperature
Hot Case	30kW	Albedo + Earth IR input	-26 °C
Cold Case	10kW	No input	-91 °C

Based on the power consumption of each operation mode (see 2.3.1.Power Modes), 30 kW is assumed for the heat input from the space station equipment as a hot case, 10 kW is assumed for the heat input from the space station equipment as a cold case. For the hot case, it is possible to keep the radiator temperature around -26 °C, which is sufficiently low for the ammonia loop to transfer the heat. On the other hand, under the cold case, ammonia temperature reaches below its freezing temperature under the Cold

Case condition. In order to keep the functionality of the radiator, counter measures are required, such as:

- Preheating of the coolant before the radiator
- Rotation control of radiator to allow heat input from Sun or Earth
- Dedicated pipe design on the radiator to allow partial coolant freezing

12.3. Options and Trades

12.3.1. One-phase or Two-phase Mechanically Pumped Fluid Loop

Pumped fluid loops can be categorized as single-phase fluid loops, or two-phase fluid loops. In single-phase fluid loops, only sensible enthalpy of circulating fluid is used for heat transfer. On the other hand, two-phase fluid loops utilize latent heat of vaporization. Advantages and disadvantages of each system are summarized in Table 12-2.

Table 12-2: Trade between one-phase and two-phase systems.

	One-phase System	Two-phase System
Advantages	Simple and well understood, Easy to test Relatively inexpensive Low risk	Large temperature drops from equipment to radiator Small mass flow rate Small pumping power
Disadvantages	Small temperature drops from equipment to radiator Large mass flow rate Large pumping power	Difficult to design Not many flight experience - TPFLEX (Two-Phase Fluid Loop Experiment) - AMS-02 (Alpha Magnetic Spectrometer)

In the last decades a large amount of work has been devoted to two-phase fluid loop system development, and several in-orbit demonstrators have been flown to check their performances in real space conditions. However, still one-phase fluid loop system would be the primary option for a manned mission which requires high reliability for every system.

12.3.2. Working Fluids (One-phase system)

For one-phase fluid system with non-expendable coolants, the following variables are the basic parameters for the fluid selection: density, specific heat, thermal conductivity, viscosity, and temperature limits. In addition, toxic/non-toxic is an important factor for a manned mission. There are several candidates for coolant fluids. However, the list of fluids already selected for single-phase fluid system is very limited. Some of the coolant fluids with flight experience are shown in Table 12-3 [RD 17].

Table 12-3: Working fluids for one-phase systems.

Working Fluid	Temperature Range	Space Application
Water, H_2O		ISS
Ammonia, NH_3	-70°C ~ +60°C	ISS
Freon-21, $CHFCl_2$		Space Shuttle
Freon-11, CCl_3F	-80°C ~ +50°C	Mars Pathfinder, Mars Exploration Rover
Galden ZT 85	-10°C ~ +70°C	Alphabus

Considering the development and operation experience in ISS, ammonia for the external fluid loop and water for the internal fluid loop would be the primary option for the fluid selection.

12.4. Mass and Power Budget

Mass and power thermal subsystem components are estimated or scaled based on the ISS Thermal Control System component information [RD 13].

12.4.1. List of Equipment

Table 12-4: Mass budget of the thermal equipment.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Ammonia_Coolant_Pipe	200.00	20.00	40.00	240.00	4.14
DN_Heat_Exchanger	80.00	20.00	16.00	96.00	1.66
DN_Heater	10.00	20.00	2.00	12.00	0.21
DN_Heater_Controller	60.00	20.00	12.00	72.00	1.24
DN_Internal_Cold_Plate	20.00	20.00	4.00	24.00	0.41
DN_MLI	70.00	20.00	14.00	84.00	1.45
DN_Water_Coolant_Pipe	20.00	20.00	4.00	24.00	0.41
DN_Water_Pump_Module	200.00	20.00	40.00	240.00	4.14
HA_Heat_Exchanger	80.00	20.00	16.00	96.00	1.66
HA_Heater	10.00	20.00	2.00	12.00	0.21
HA_Heater_Controller	60.00	20.00	12.00	72.00	1.24
HA_Internal_Cold_Plate	20.00	20.00	4.00	24.00	0.41
HA_MLI	250.00	20.00	50.00	300.00	5.18
HA_Water_Coolant_Pipe	100.00	20.00	20.00	120.00	2.07
HA_Water_Pump_Module	200.00	20.00	40.00	240.00	4.14
SM_Ammonia_Pump_Module	700.00	20.00	140.00	840.00	14.49
SM_Ammonia_Tank_Assembly	500.00	20.00	100.00	600.00	10.35
SM_External_Cold_Plate	100.00	20.00	20.00	120.00	2.07
SM_Heat_Exchanger	80.00	20.00	16.00	96.00	1.66
SM_Heater	10.00	20.00	2.00	12.00	0.21
SM_Heater_Controller	60.00	20.00	12.00	72.00	1.24
SM_Internal_Cold_Plate	20.00	20.00	4.00	24.00	0.41
SM_MLI	80.00	20.00	16.00	96.00	1.66
SM_Nitrogen_Tank_Assembly	400.00	20.00	80.00	480.00	8.28
SM_Radiator	1280.00	20.00	256.00	1536.00	26.50
SM_Water_Coolant_Pipe	20.00	20.00	4.00	24.00	0.41
SM_Water_Pump_Module	200.00	20.00	40.00	240.00	4.14

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	4830.00			5796.00	

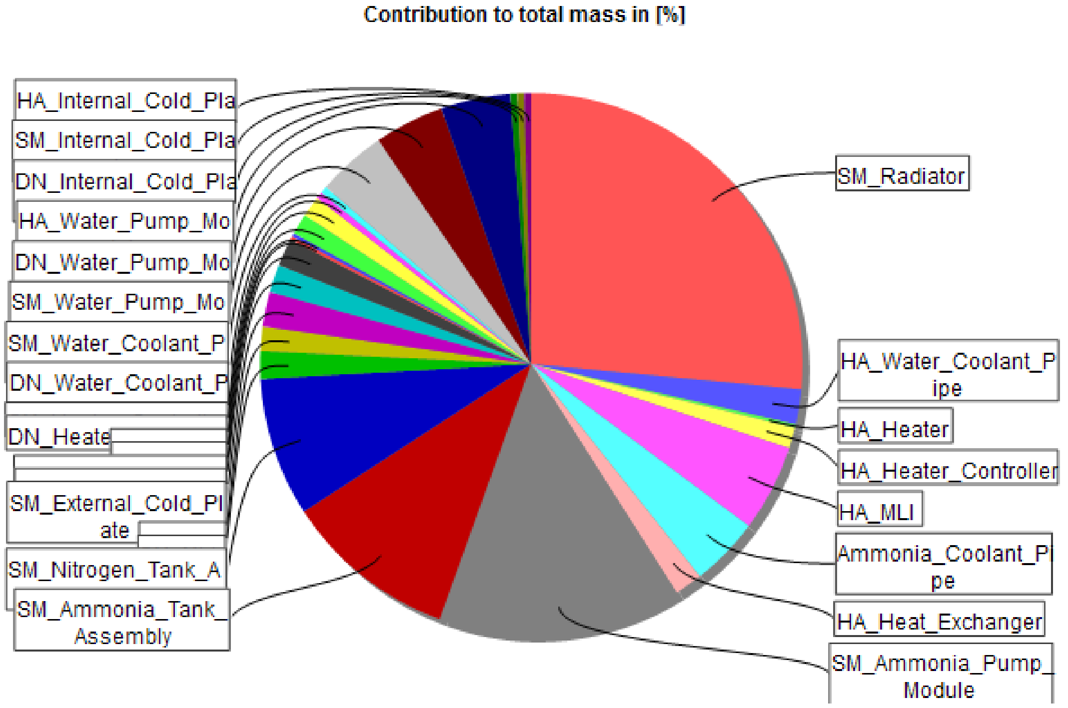


Figure 12-4: Pie chart of mass distribution of the thermal equipment

12.4.2. Power Budget

Table 12-5: Power budget of the thermal subsystem.

System Component	Parameter	Unit	Default	Standard_elect...	CrewExchange	Docked	Survival	Proximity_Ops
Thermal	power_avg_wMargin	Watt	2801.280	2801.280	2801.280	2801.280	2801.280	2801.280
Ammonia_Coolant_Pipe	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
DN_MLI	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
DN_Heater	power_avg_wMargin	Watt	540.000	540.000	540.000	540.000	540.000	540.000
DN_Heater_Controller	power_avg_wMargin	Watt	19.680	19.680	19.680	19.680	19.680	19.680
DN_Heat_Exchanger	power_avg_wMargin	Watt	20.160	20.160	20.160	20.160	20.160	20.160
DN_Water_Coolant_Pipe	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
DN_Water_Pump_Module	power_avg_wMargin	Watt	192.960	192.960	192.960	192.960	192.960	192.960
HA_MLI	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
HA_Heater	power_avg_wMargin	Watt	540.000	540.000	540.000	540.000	540.000	540.000
HA_Heater_Controller	power_avg_wMargin	Watt	19.680	19.680	19.680	19.680	19.680	19.680
HA_Heat_Exchanger	power_avg_wMargin	Watt	20.160	20.160	20.160	20.160	20.160	20.160
HA_Water_Coolant_Pipe	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
HA_Water_Pump_Module	power_avg_wMargin	Watt	192.960	192.960	192.960	192.960	192.960	192.960
SM_MLI	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
SM_Heater	power_avg_wMargin	Watt	540.000	540.000	540.000	540.000	540.000	540.000
SM_Heater_Controller	power_avg_wMargin	Watt	19.680	19.680	19.680	19.680	19.680	19.680
SM_Heat_Exchanger	power_avg_wMargin	Watt	20.160	20.160	20.160	20.160	20.160	20.160
SM_Water_Coolant_Pipe	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
SM_Water_Pump_Module	power_avg_wMargin	Watt	192.960	192.960	192.960	192.960	192.960	192.960
SM_Ammonia_Pump_Module	power_avg_wMargin	Watt	442.560	442.560	442.560	442.560	442.560	442.560
SM_Ammonia_Tank_Assembl	power_avg_wMargin	Watt	20.160	20.160	20.160	20.160	20.160	20.160
SM_External_Cold_Plate	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
SM_Nitrogen_Tank_Assembly	power_avg_wMargin	Watt	20.160	20.160	20.160	20.160	20.160	20.160
SM_Radiator	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
DN_Internal_Cold_Plate	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
HA_Internal_Cold_Plate	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
SM_Internal_Cold_Plate	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
Thermal	power_energy_wMargin	Joule	3388428288.000	3388428288....	1694214144....	1694214144....	968122368.000	242030592.000

12.4.3. Mode dependencies

Thermal Control System is to be active during the entire mission phases to handle excess heat from space station equipment. Therefore, Thermal Control System does not change the operation scheme regardless of the system operation mode. On the other hand, it is expected that the power consumption of the Thermal Control System differs depending on the operation mode and environmental heat input, because of the different necessity of heater power, radiator rotation control and so on. In order to estimate such power consumption, it is necessary to have detailed design for space station equipment configuration and heater allocations.

12.5. Re-Supply Items

- Preservative for Cooling Water
  - Ex. Ortho-Phthalaldehyde
  - Every 2~3 years

12.6. To Be Further Studied / Additional Considerations

For the future study, the following points are considered as important topics:

- Trade-off study and detailed design of fluid loop
    - Fluid type trade-off
    - Fluid pipe routing design
    - Positioning of heat exchangers / cold plates
    - Fluid control (valve) system design
- Thermal Modelling and Simulation
    - Hot operation/environment case
    - Cold operation/environment case
    - Transient case such as special attitude control operation or short term high power operation
- Configuration update and heater operation design considering temperature limit of each equipment

12.7. Summary

In this study, conceptual design of the space station Thermal Control System was performed. Based on the heritages from ISS design, development and operation, baseline system architecture was proposed with possible optional designs, and the mass power budget of baseline system was evaluated. Within the radiator feasibility study, it is shown that the radiator with the external fluid loop is able to cover the expected range of heat generation in different operation modes.

13. Structure

13.1. Requirements and Design Drivers

- Length of each module fixed in previous studies [RD 25].

Table 13-1: First mass and size budget estimation [RD 25].

Module	Size Estimate in m	Mass Estimate in t
Free-Flyer	launch configuration: Ø=4.5; length=9.8	20
External Science Platform	*length=7; width=7; height=1 (deployed)*	*3.2*
Pressurized Laboratory	*Ø=4.5; length=3.4*	*6.9*
Service Module	*Ø=4.5; length=5.4*	*9.9*
Docking Node	Ø=4.5; length=6.7	15.5
Habitat + Laboratory	Ø=7.5; length=13.7 (expanded)	21
Service Module	Ø=4.5; length=5.4	9.9
EVA Module	Ø=2.55; length=4.9	3.6

- Human Spaceflight: Mission Analysis and Design [RD 27]. Guide for modules mass estimation 1.6 t/m.
    - Docking Node

$$1.6\,\frac{tn}{m} * 6.7\,m = 10.7\,t$$

    - Service Module

$$1.6\,\frac{tn}{m} * 5.4\,m = 8.6\,t$$

- Equipment mass must be subtracted from above calculated values.
- Habitat module Bigelow B330. Mass procured by company [RD 22].
- IBDM as reference for docking adapters [RD 23].
- Corridor between SM/DN and CMG structure estimated as simple geometrical aluminium shapes.
- Solar Array structure modelled following [RD 26]. Rectangular tubes.
- Similar Cupola as ISS [RD 24].

13.2. Baseline Design

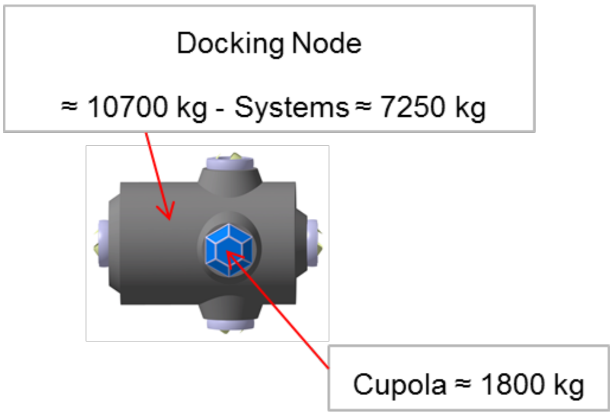


Figure 13-1: Mass assumptions for the Docking Node.

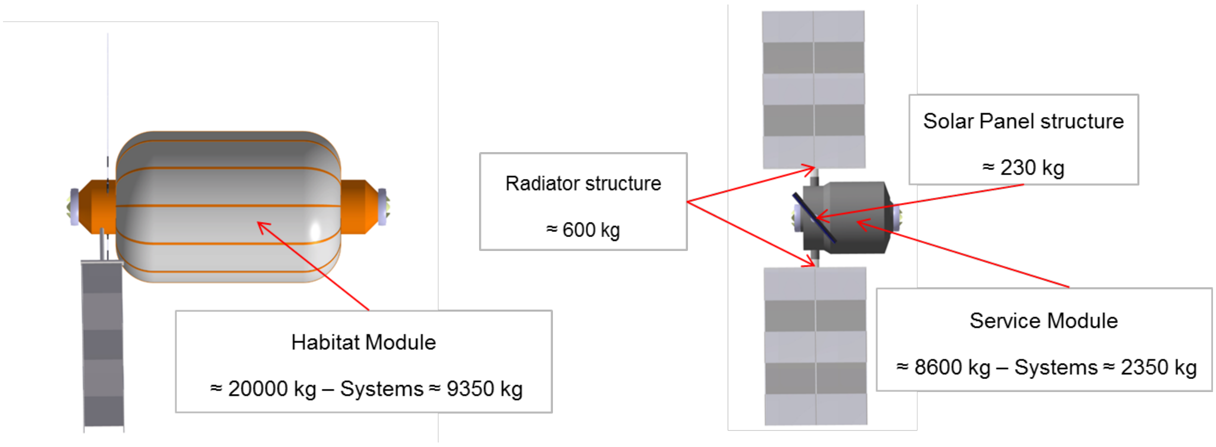


Figure 13-2: Mass assumptions for the Habitat and Service Module.

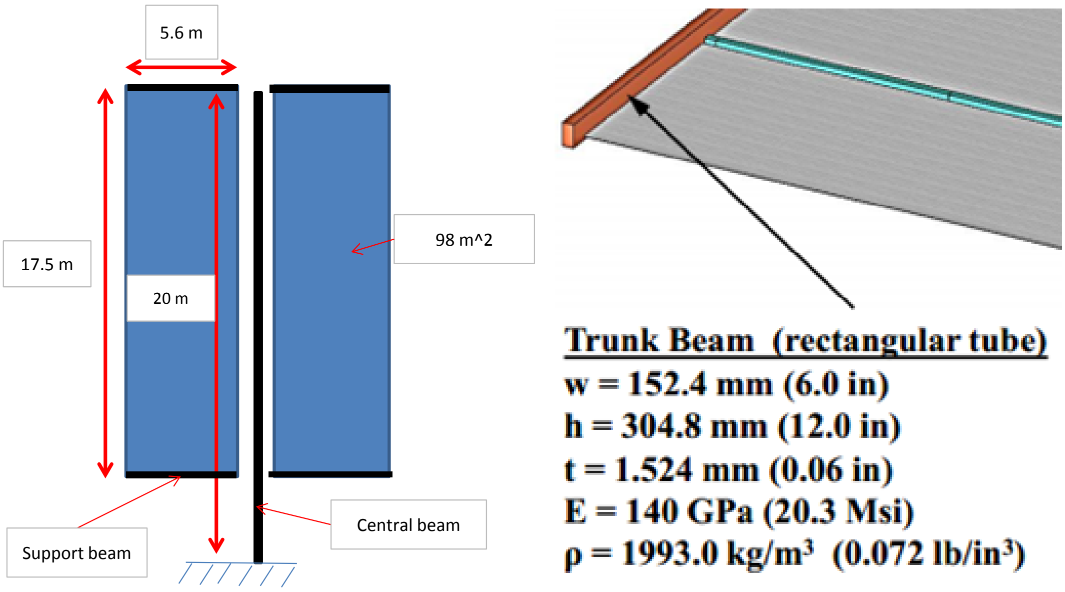


Figure 13-3: Simplified design of solar panel structure. Beam model source [RD 26].

Table 13-2: Solar panel structure mass assumption.

	Support Beam	Central Beam
w	0.15 m	0.15 m
h	0.30 m	0.30 m
t	0.0015 m	0.0015 m
L	5.6 m	20 m
ρ	2 000 kg /m^3	2 000 kg /m^3
Mass_unit	15 kg	54 kg
Units	8	2
Mass_Total	121 kg	108 kg

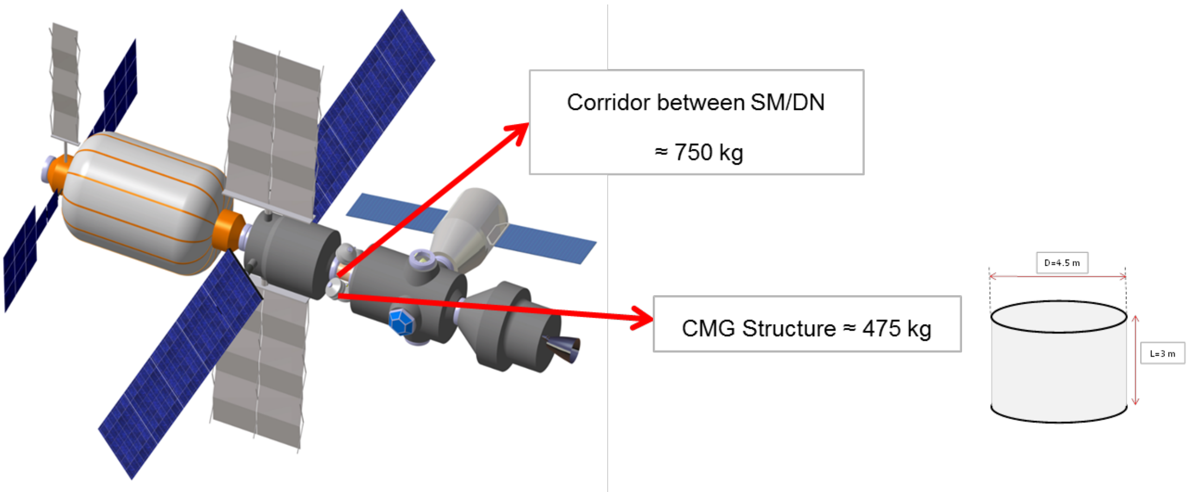
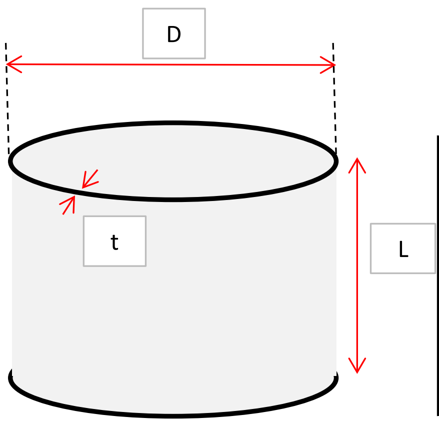


Figure 13-4: Mass assumptions for the CMG construct.

Table 13-3: Corridor and CMG mass assumption.



	Corridor	CMG Structure
L	1 m	3.0 m
t	0.08 m	0.004 m
D	1.06 m	4.5 m
ρ	2 800 kg /m^3	2 800 kg /m^3
Mass_unit	746 kg	475 kg
Units	1	1
Mass_Total	746 kg	475 kg

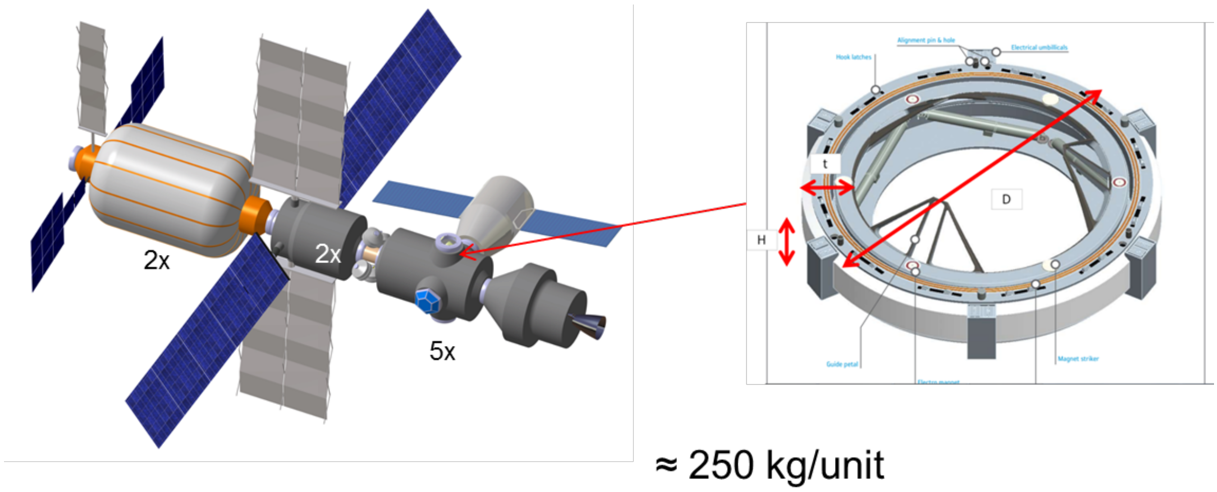


Figure 13-5: Mass assumptions for the docking adapters.

Table 13-4: Docking adapter mass assumption.

D	1.50 m
t	0.30 m
H	0.40 m
ρ	2 800 kg /m^3
Parameter	2
Mass_unit	252 kg
Units	9
Mass_Total	2268 kg

13.3. Options and Trades

- Lighter Cupola options.

13.4. Mass Budget

13.4.1. List of Equipment

Table 13-5: Mass budget of the structure domain.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
SM_Solar_panel_structure	456.00	20.00	91.20	547.20	1.83
SM_ServiceModule	2366.95	20.00	473.39	2840.34	9.49
SM_Radiator_support_structure	600.00	20.00	120.00	720.00	2.41
SM_Docking_adapter	504.00	20.00	100.80	604.80	2.02
SM_CMG_Structure	475.00	100.00	0.00	475.00	1.59
HA_Habitat	9354.87	20.00	1870.97	11225.84	37.52
HA_Docking_adapter	504.00	20.00	100.80	604.80	2.02
DN_DockingNode	7246.10	20.00	1449.22	8695.32	29.06
DN_Docking_adapter	1260.00	20.00	252.00	1512.00	5.05
DN_Cupola	1800.00	100.00	0.00	1800.00	6.02
Connection_corridor	746.00	20.00	149.20	895.20	2.99
Component	0.00	CalcErr	0.00	0.00	0.00
	Mass w/o margin [kg]	**Margin [%]**	**Margin [kg]**	**Mass with margin [kg]**	**% of total dry mass**
Total dry mass:	25312.92			29920.50	

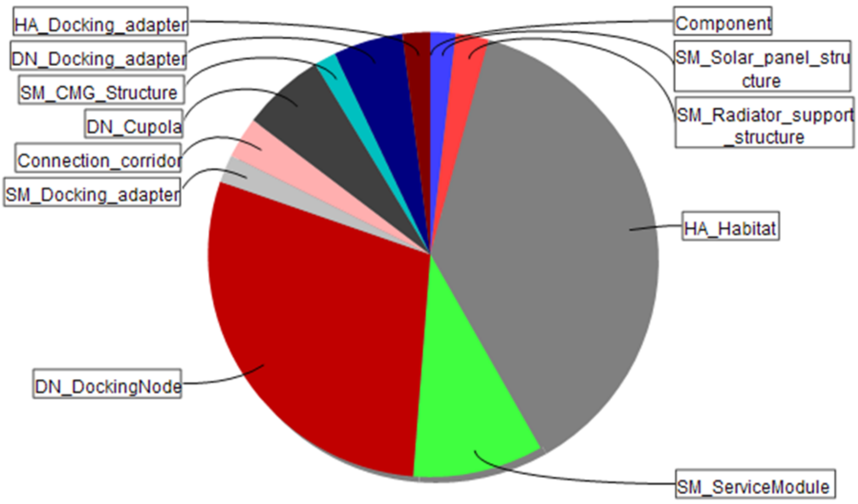


Figure 13-6: Pie chart of mass distribution of the structure parts.

13.4.2. Power Budget

There are no structural parts which require power.

13.5. To Be Further Studied / Additional Considerations

- HA_B330
- ECLSS System mass redistribution between DN, SM and HA
- Refine CMG structure, Corridor structure.
- Recalculate solar panel structure when its design is frozen.
- Take into account new available materials into the mass estimations.

14. Robotics

This section describes robotic payload on the planned station. This comprises a robotic manipulator mounted externally for experiment operations on the free-flyer as well as a preliminary collection of potential robotic experiments on-board and outside of the station.

14.1. Requirements and Design Drivers

Deriving from the chosen configuration of the station modules, the following requirements and high-level technical specifications of the arm were identified:

- The arm shall be used to pick experiments from the airlock and place them on the experiment platform. The way experiments are mounted is still to be defined and will be assessed in more detail when studying the free-flyer.
- There will be no additional arm for capsule or other vehicle berthing to the station; docking shall be executed autonomously
- Looking at the geometric necessities, approx. 7 m need to be bridged by the arm (lab + hab end cap + grasp into airlock + height difference between platform and hab).
- In order to optimize the available workspace with respect to the experiment platform, a linear rail shall be used. Consequently, a shorter arm with connected increased precision, lower weight etc. can be used.

14.2. Baseline Design

Figure 14-1 depicts the major geometric specifications relevant to the robotic arm and the location of the linear rail on the experiment platform. Basically the arm represents the interface between the free-flyer and base station with respect to experiment hand-over. It must be able to robustly grasp the experiment in the air lock and place it at an arbitrary position on the experiment platform. To achieve this, both the laboratory as well as the end cap of the habitat have to be bridged. In addition the manipulator has to grasp into the air lock and additionally account for the difference in height between experiment platform and air lock.

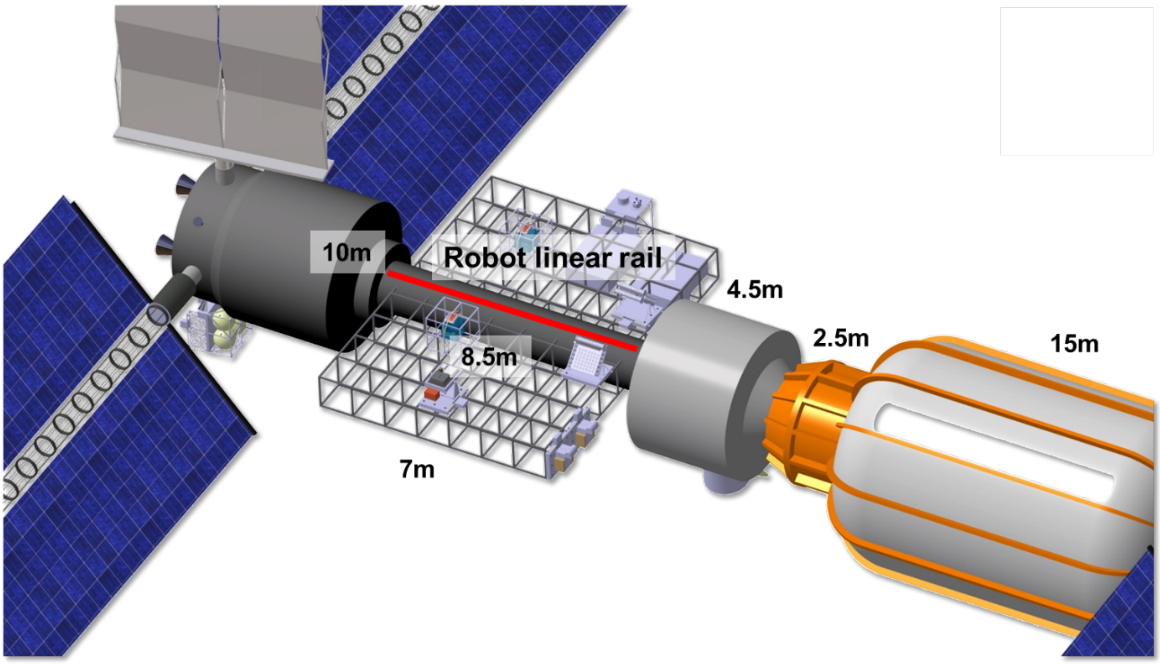


Figure 14-1: Baseline mounting of the robotic arm.

Following this geometric situation and good engineering practice in robotics, the following baseline design for the kinematics and technical specifications of the arm was defined:

- Total length of approx. 10m, approx. half the length in stowed configuration
- 7 degrees of freedom (DoF) in alternating roll-pitch configurations
- Stereo-camera system at arm wrist for visual servoing in order to increase pointing accuracy
- Max. nominal torque of 160Nm due to currently available, space-qualified robotic joints
- Integrated joint design featuring position, torque sensor and break allowing high-precision position and force-sensitive impedance control for compliant and robust grasp operations
- Redundant mechatronic design
- The arm can be controlled in autonomous, shared autonomy and telepresence mode

Figure 14-2 depicts a rendering of such a 7-DoF arm configuration with a gripper and a wrist camera system attached. Please note that the depicted link lengths do not fit the specifications as outlined above. However, the shown geometric relations are very similar to what a station arm could look like. The arm is a very long and relatively thin structure that possesses intrinsic mechanical flexibility. In order to increase pointing accuracy of the arm's end-effector, a stereo-camera system is utilized for model-based visual servoing. That way, the arm is guided to its target, cancelling out possible uncertainties and inaccuracies. In the addition, the integrated joint design comprising position and torque sensors as well as a motor break allows a fine-tuned and force-sensitive control approach for detecting collisions and reacting in robust manner to it without jeopardizing the payload and the arm itself.

Table 14-1 and Table 14-2 list the respective figures for the arm kinematics describing the geometric configuration as mentioned above. The subsequent free-flyer study will go into more detail and present detailed kinematics analysis taking into account self-collision and collision with the space station structure the manipulator is mounted onto.

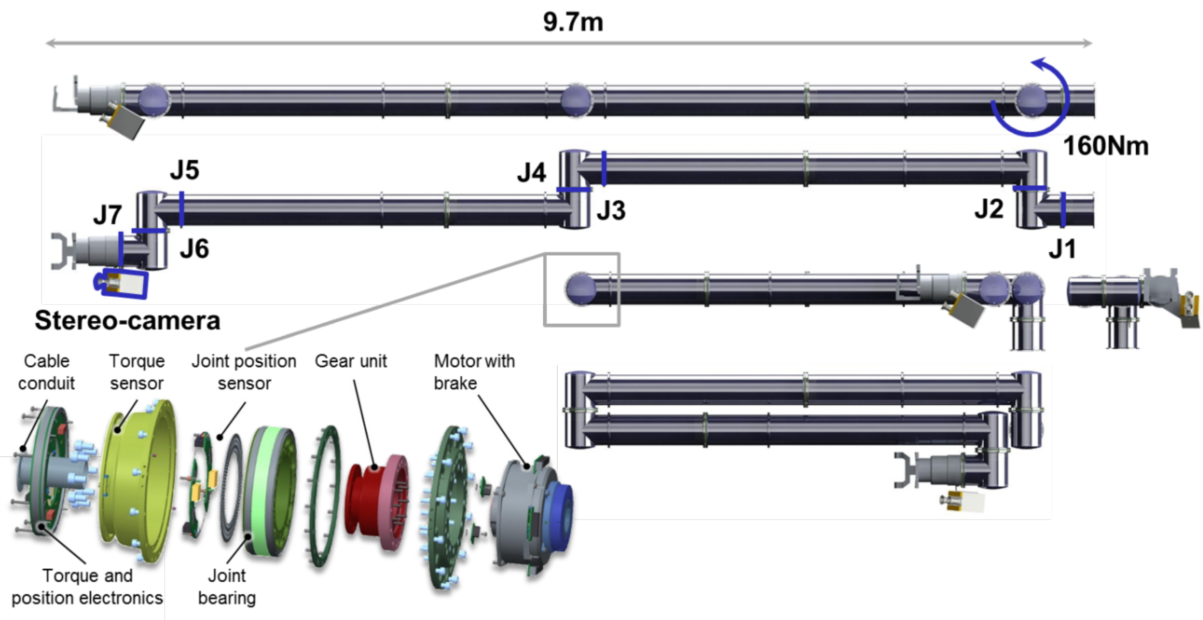


Figure 14-2: Robotic arm configuration.

Table 14-1: Robotic arm kinematics.

total arm length [m]			**9,706**	
long segments difference [m]			**0,140**	
a [m]	**alpha [deg]**	**theta [deg]**	**d [m]**	**type**
0	0	0	1,756	roll
0	-90	0	0,168	pitch
0	90	180	3,870	roll
0	-90	0	0,168	pitch
0	90	180	3,730	roll
0	90	0	0,168	pitch
0	-90	0	0,350	roll

Table 14-2: Degree of freedom of joints of robotic arm.

Joint	Lower limit [deg]	Upper limit [deg]	Zero [deg]	Stowed [deg]	Approach Init [deg]
#1	-158	158	0,000	0	0
#2	-158	158	0,000	-90	-45
#3	-158	158	0,000	0	0
#4	-188	128	0,000	-180	-50
#5	-158	158	0,000	0	0
#6	-128	188	0,000	180	-90
#7	-158	158	0,000	0	0

14.3. Options and Trades

The mass, volume and power values of the following experiments are covered by the Science / Payload domain (section 4.2.3, page 38).

In addition to the robotic manipulator, two potential robotic experiments have been proposed, partly orienting on currently ongoing activities on the International Space Station (ISS) that could be extended in the future.

14.3.1. NanoSat Freeflyers

NanoSat freeflyers are currently being used an experiment platform aboard ISS in the form of the SPHERES experimental setup. Here, three nanosat-sized satellites, propelled by cold gas execute control experiments investigating aspects of swarm flight, docking, visual navigation and human-machine interaction amongst others. Currently these satellites are only built to fly inside of the habitat only.

In order to increase the operational capabilities, one could go one step further from solely being an experiment to also supporting extra-vehicular activities (EVA) by delivering additional spatial perspectives and to giving the possibility for checking the integrity of the stations outer hull without the need for an EVA. For that, such an extended SPHERES experiment and actual operation equipment of a new station would have to be made ready for space flight, including extended thermal and power subsystem etc.

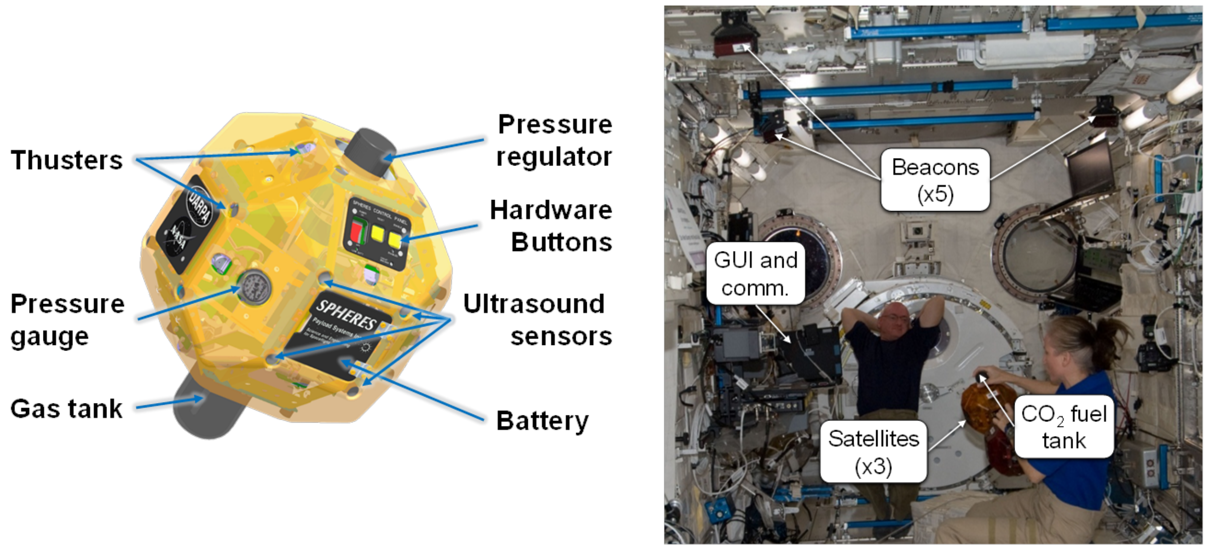


Figure 14-3: NanoSat Freeflyer configuration (here: SPHERES).

That way, this fault-tolerant platform could be increased in its reach going from computer simulation all the way up to real flight testing in space, cp. Figure 14-4. This could improve the way space flight algorithms, e.g. for navigation, docking and swarm flight, are developed by increasing their performance and actually flight-testing them without risking a loss of the complete mission.

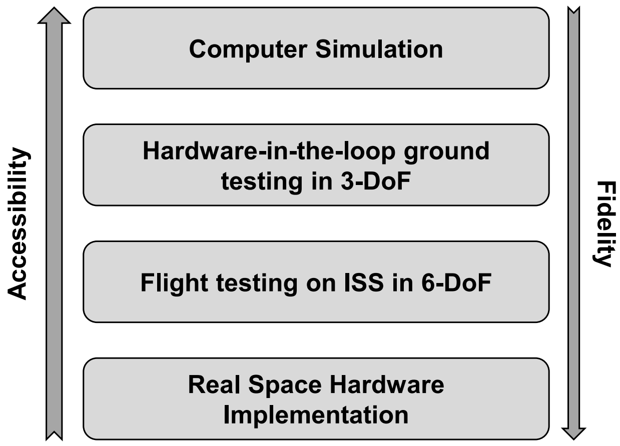


Figure 14-4: NanoSat's development roadmap.

14.3.2. Robotic Assistant

As a second experiment, a robotic assistant was proposes, cp. Figure 14-5. Such a human body-like upper torso connected to some type of mobile base could be used both for internal testing and experimentation, e.g. with respect to different control model, and for external operations, e.g. maintenance & repairs or EVA support operations.

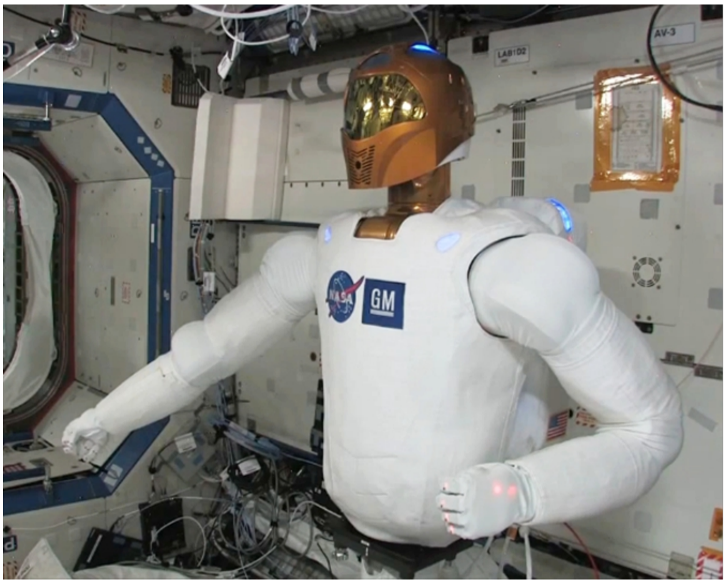



Figure 14-5: Robotic assistant (left: NASA Robonaut; right: DLR Space Justin).

Figure 14-6 shows an exemplary schematic setup for the telepresent control of such a humanoid robotic assistant. The operator on ground or aboard the station received multimodal feedback from the teleoperator and can actively interact with the remote environment. In addition to telepresence, other control modes ranging from fully autonomous operations up to supervised and shared control approach are thinkable. In this regard, a lot of potential experimental setups are thinkable including on-orbit servicing operations such as repairing and refuelling a defunctive satellite.

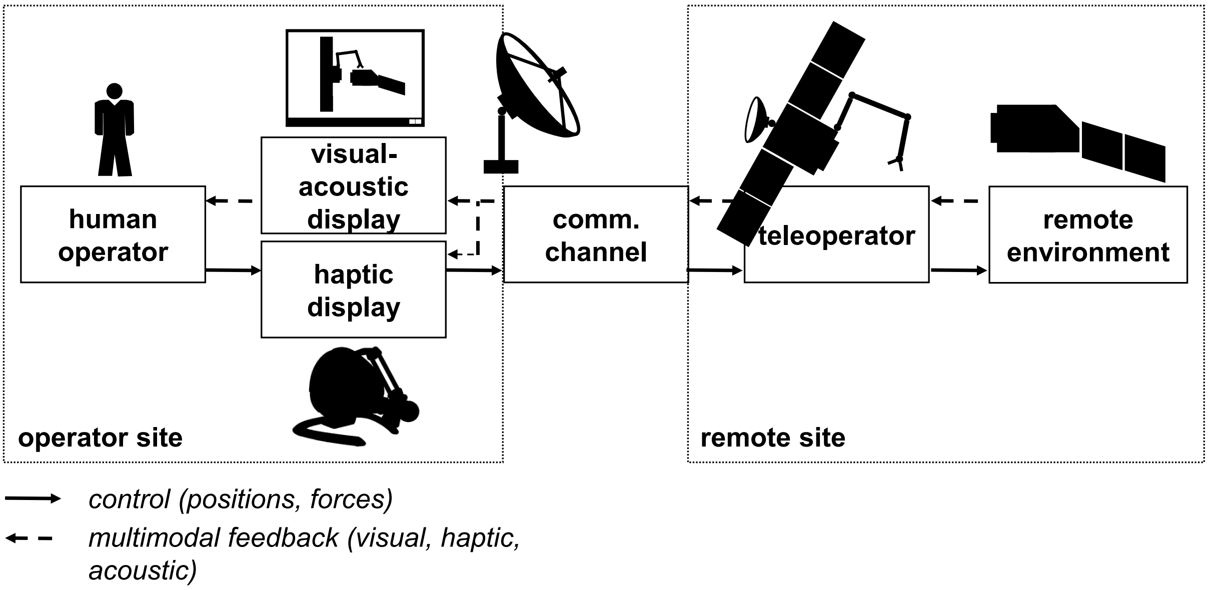


Figure 14-6: Control modes for robotic assistance.

14.4. Mass and Power Budget

This chapter gives a brief overview of the preliminary mass and power budget for the introduced robotic mechanisms and experiments.

14.4.1. List of Equipment

Table 14-3: Mass budget of the robotic arm.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Experiment_Manipulator	117.00	20.00	23.40	140.40	92.13
RCU	10.00	20.00	2.00	12.00	7.87

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	127.00			152.40	

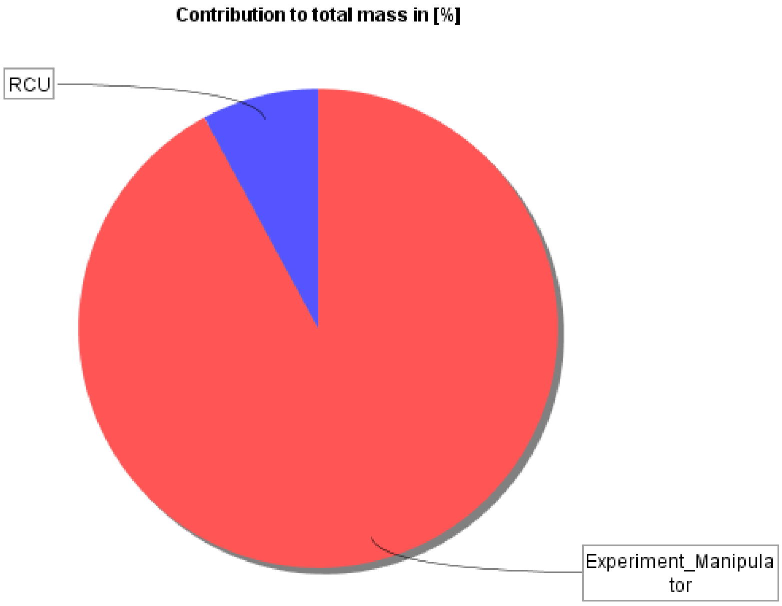


Figure 14-7: Pie chart of mass distribution of the robotic arm.

14.4.2. Power Budget

Table 14-4: Power budget of the robotic arm.

System Component	Parameter	Unit	Default	Standard_elect...	CrewExchange	Docked	Survival	Proximity_Ops
Robotic_Mechanisms	power_avg_wMargin	Watt	27.000	27.000	27.000	27.000	0.000	27.000
Experiment_Manipulator	power_avg_wMargin	Watt	18.000	18.000	18.000	18.000	0.000	18.000
RCU	power_avg_wMargin	Watt	9.000	9.000	9.000	9.000	0.000	9.000
Robotic_Mechanisms	power_energy_wMargin	Joule	32659200.000	32659200.000	16329600.000	16329600.000	0.000	2332800.000

14.4.3. Mode dependencies

There are no major mode dependencies besides that during safe mode, both the manipulator and the experiments shall be completely powered down.

14.5. Re-Supply Items

For the currently proposed experiments, re-supply items would be batteries and cold gas tanks. By using rechargeable accumulators, re-supply would be limited to tanks only. The amount depends on the experiment schedule and would be roughly 50 tanks per year.

14.6. To Be Further Studied / Additional Considerations

The following points have to be further assessed in the scope of the subsequent free-flyer study:

- Mechanism to fix experiments
- Required gripper mechanics for the robotic manipulator
- Detailed reachability analysis
- Launch configuration and stowage

14.7. Summary

This chapter presented the required robotic payload for the space station and additionally two robotic experiments to be used both internally and externally. Both experiments, the nanosat free-flyers as well as the robotic assistant could be used for scientific experimentation and actual operational work on the station, effectively limiting the need for astronaut EVA’s.

15. AOCS

15.1. Requirements and Design Drivers

- Station is oriented for balancing drag torque (in all modes except PoM).
- Nominal attitude:
    - DN in flight direction
    - HA in flight direction for debris avoidance if no vehicle is docked
    - If the visiting vehicle is docked on the DN the attitude needs to be HA in flight direction for orbit raising, optionally the visiting vehicle is docked to the HA and FF docks to DN (only if needed)
- For assembly HA+FF is chaser, SM+DN is target.
- RVD-Sensors are inactive after assembly.
- Sensors and actuators are distributed on both modules (HA and SM+DN).
- Sensors and actuators are redundant on both modules (HA and SM+DN).
- Some sensors are not usable due to obstructed FOV after assembly or when visiting vehicles are docked. → Deactivated
- Design shall be tolerant to at least two failures in assembled configuration.

15.2. Baseline Design

- Service Module + Docking Node
    - CMG assembly (SM)
    - Rate Gyro Assembly (SM)
    - Star Tracker (SM top side)
    - Horizon Sensor (bottom side SM)
    - GNSS Receiver + Antenna (on DN far end to avoid FOV obstruction)
    - Sun Sensor (on DN far end to avoid FOV obstruction)
    - Thrusters only on DN end, attitude and orbit capability. The orbit capability is a contingency to allow orbit raising or space debris avoidance manoeuvres in case no vehicle with available propulsion is docked.

- Habitat Module
    - IMU + RVDS (only for first assembly)
    - GNSS Receiver + Antenna
    - Sun Sensor (on far end towards FF docking port to avoid FOV obstruction)
    - Horizon Sensor (bottom side)
    - Thrusters (on both ends, attitude capability)

15.3. Options and Trades

15.3.1. SM active part instead of FF

- Discarded option: HA is target and docked by SM during assembly.
    - CMGs remain on SM.
    - Thrusters with full attitude capability are needed on HA (on both ends).
    - Thrusters with attitude and translational capability needed on SM.

    → More thrusters needed in total.

- Docking equipment (RVDS + IMU) is only one-failure-tolerant.
    - Similar to ATV.

15.3.2. SM and DN launched separate

If Docking Node and Service Module had to be launched in two pieces, then the modules need to following equipment:

- SM:
    - CMG assembly
    - Rate Gyro Assembly
    - Star Tracker (top side)
    - Horizon Sensor (bottom side)
    - GNSS Receiver + Antenna (on SM far end to avoid FOV obstruction)
    - Sun Sensor (on SM far end to avoid FOV obstruction)
    - Thrusters on SM (only one end, attitude capability together with CMGs)

- for DN:
    - IMU* + RVDS* (only for first assembly)
    - GNSS Receiver* + Antenna*
    - Sun Sensor* (on far end to avoid FOV obstruction)
    - Horizon Sensor* (bottom side)
    - Thrusters* on both ends, attitude and translational capability
    - Orbit raising thrusters (on aft as contingency)

All equipment marked with * has to be added on top of the current baseline design.

15.4. Mass and Power Budget

15.4.1. List of Equipment

The current design is based on existing equipment which can be improved by delta developments. The selected CMGs are used on the ISS. Since the designed station is smaller a re-design of the CMGs is expected to reduce mass and power consumption of the CMGs.

For GNSS receiver and antenna technology European state-of-the-art is assumed. A delta development might be needed to allow high precision differential GNSS observations for gaining attitude information and high precision relative position to visiting vehicles and the FF during rendezvous and docking.

Table 15-1: Mass budget of the AOCS equipment.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
DN_GNSS_Receiver_Antenna	1.20	20.00	0.24	1.44	0.11
DN_Horizon_Sensor	7.00	5.00	0.35	7.35	0.57
DN_Sun_Sensor	1.80	5.00	0.09	1.89	0.15
HA_GNSS_Receiver	12.00	20.00	2.40	14.40	1.12
HA_GNSS_Receiver_Antenna	1.20	20.00	0.24	1.44	0.11
HA_Horizon_Sensor	7.00	5.00	0.35	7.35	0.57
HA_Rendezvous_Docking_Sensor	16.00	20.00	3.20	19.20	1.49
HA_Rendezvous_IMU	1.50	5.00	0.07	1.57	0.12
HA_Sun_Sensor	1.80	5.00	0.09	1.89	0.15
SM_CMG	1088.00	10.00	108.80	1196.80	93.09
SM_GNSS_Receiver	12.00	20.00	2.40	14.40	1.12
SM_Rate_Gyro_Assembly	8.40	5.00	0.42	8.82	0.69
SM_Star_Tracker	8.70	5.00	0.44	9.14	0.71

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
Total dry mass:	1166.60			1285.69	

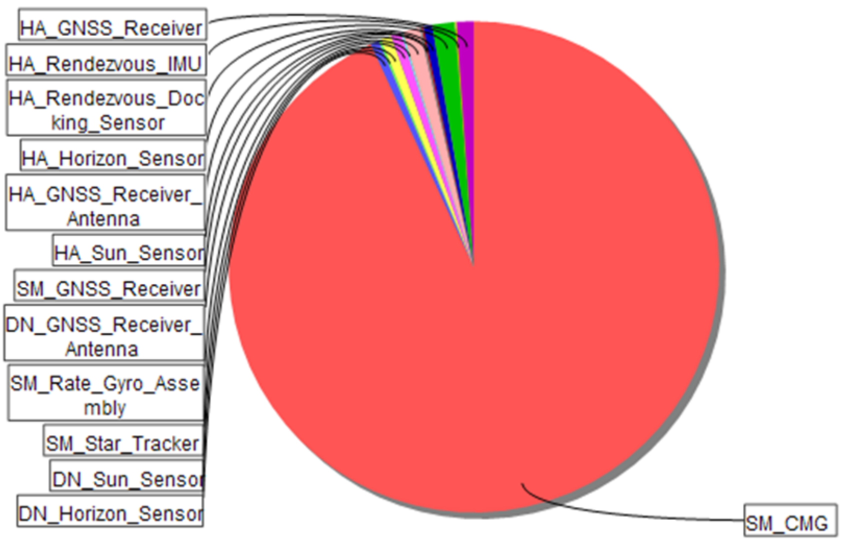


Figure 15-1: Pie chart of mass distribution of the AOCS equipment.

15.4.2. Power Budget

Table 15-2: Power budget of the AOCS equipment.

System Component	Parameter	Unit	Default	Standard_elect...	CrewExchange	Docked	Survival	Proximity_Ops
AOCS	power_avg_wMargin	Watt	286.495	286.495	286.495	286.495	255.516	298.815
DN_GNSS_Receiver_Antenna	power_avg_wMargin	Watt	1.200	1.200	1.200	1.200	1.200	1.200
DN_Horizon_Sensor	power_avg_wMargin	Watt	7.875	7.875	7.875	7.875	0.000	7.875
DN_Sun_Sensor	power_avg_wMargin	Watt	3.150	3.150	3.150	3.150	2.098	3.150
HA_GNSS_Receiver_Antenna	power_avg_wMargin	Watt	1.200	1.200	1.200	1.200	1.200	1.200
HA_Horizon_Sensor	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
HA_Rendezvous_Docking_Ser	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
HA_Rendezvous_IMU	power_avg_wMargin	Watt	0.000	0.000	0.000	0.000	0.000	0.000
HA_Sun_Sensor	power_avg_wMargin	Watt	3.150	3.150	3.150	3.150	2.098	3.150
SM_CMG	power_avg_wMargin	Watt	188.320	188.320	188.320	188.320	188.320	200.640
SM_GNSS_Receiver	power_avg_wMargin	Watt	24.000	24.000	24.000	24.000	24.000	24.000
SM_Rate_Gyro_Assembly	power_avg_wMargin	Watt	12.600	12.600	12.600	12.600	12.600	12.600
SM_Star_Tracker	power_avg_wMargin	Watt	21.000	21.000	21.000	21.000	0.000	21.000
HA_GNSS_Receiver	power_avg_wMargin	Watt	24.000	24.000	24.000	24.000	24.000	24.000
AOCS	power_energy_wMargin	Joule	346544352.000	346544352.000	173272176.000	173272176.000	88306260.480	25817616.000

15.4.3. Mode dependencies

In all modes except Proximity Operations Mode the attitude of the station is oriented to balance the drag torque. In Proximity Operations Mode the attitude is horizontal to allow docking. Since the drag torque is not balanced in this attitude more power I needed by the CMGs. In Survival Mode the power consumption is reduced by switching off functionally redundant sensors. The consumption can be more reduced by switching off the CMGs at the cost to spend more fuel for attitude control.

15.5. Re-Supply Items

- Only spare parts in case of failures.

15.6. To Be Further Studied / Additional Considerations

- More detailed simulation to better estimate fuel consumption for manoeuvres and attitude stabilisation in all modes.
    - Simulation of CMG control and CMG desaturation needed.
    - More detailed disturbance modelling required.

15.7. Summary

The AOCS subsystem uses existing mature equipment which is available. No technology development is needed if U.S. technology (ITAR) is available (GNSS receiver). A re-design of the CMG would allow reducing mass and power consumption at the cost of development risk.

16. Propulsion

16.1. Requirements and Design Drivers

The propulsion system of the station is required for:

- **Orbit raising:** the drag-induced altitude loss needs to be compensated. The requirement for this is a ΔV increase of 93 m/s per year.
- **Docking:** the base station shall be assembled from either two or three modules; the habitat, service module (SM) and docking node (DN) (or the combined SM/DN). Propulsion is required by both active and passive sides.
- **Torque countering:** torque of approximately 5 Nm shall be produced by the large solar arrays. If no passive solution is found, this shall have to be countered by the propulsion system.
- **Debris avoidance:** from ISS experience, debris avoidance manoeuvres have a high ΔV requirement of 0.5 - 1 m/s per manoeuvre. The ISS has performed (on averaged, throughout its operation life) 4 such manoeuvres per year. Given the increase in debris field density and improving detection technologies, 6 manoeuvres per year shall be taken as the baseline for this study.

In order to determine the baseline propulsion system design, the critical sizing cases needed to be considered. The most demanding of these above specified needs are the debris avoidance and the orbit raising needs. The assembly of the station is the primary driver of the thruster layout, and also contributes significantly to the propellant budget.

16.2. Baseline Design

The baseline solution for the station assembly sequence already outlined in the Systems and AOCS sections of this report (Sections 2.3 and 15.2 , respectively), is as follows:

1. Free Flyer would be inserted into the desired orbit. It will be an active system.
2. Habitat. It shall be passive (Free Flyer is the chaser).
3. SM. It shall be passive (combined Free Flyer and Habitat is the chaser).
4. *DN. It shall be passive (combined Free Flyer, Habitat and SM is the chaser).* [7]
5. Crew/resupply vehicle. It shall be the chaser, and the base station shall be the target.

Following the initial station assembly, the Free Flyer shall handle all manoeuvring and propulsive requirements during the phases when it is docked. The visiting vehicle shall otherwise perform these. The SM and DN shall be able to perform boosts in contingency

[7] For case where SM and DN must be launched separately.

cases. The primary contingency involves case the phases during the relocation of the docked vehicle (Scenario Two in below figure).

The main four scenarios are outlined below. In Scenario Two of this diagram, "electrical" refers to the use of electrical propulsion for orbit-raising (outlined later in this chapter).

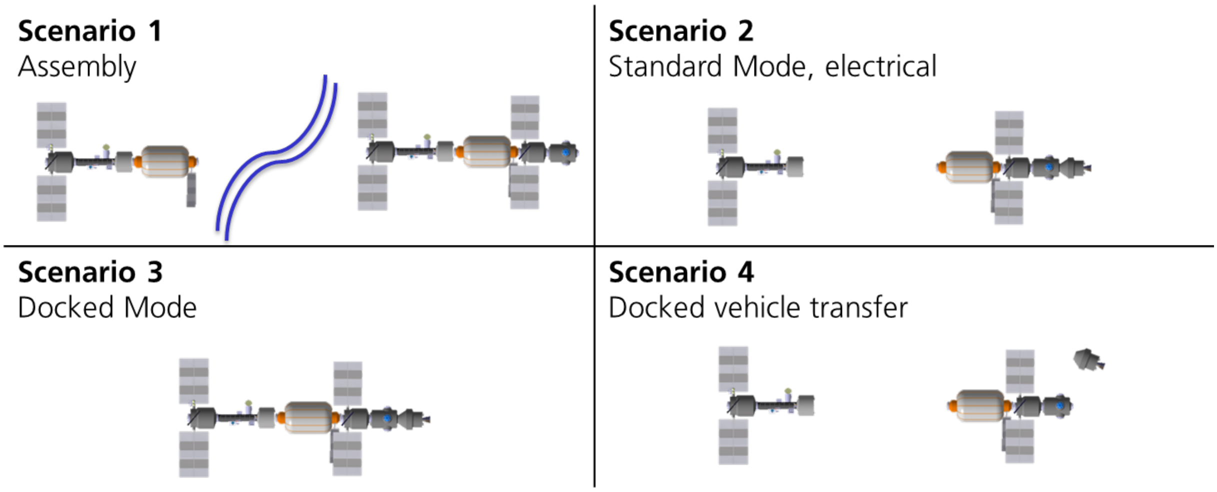


Figure 16-1: Basic scenarios during Orbital-Hub's assembly and operation.

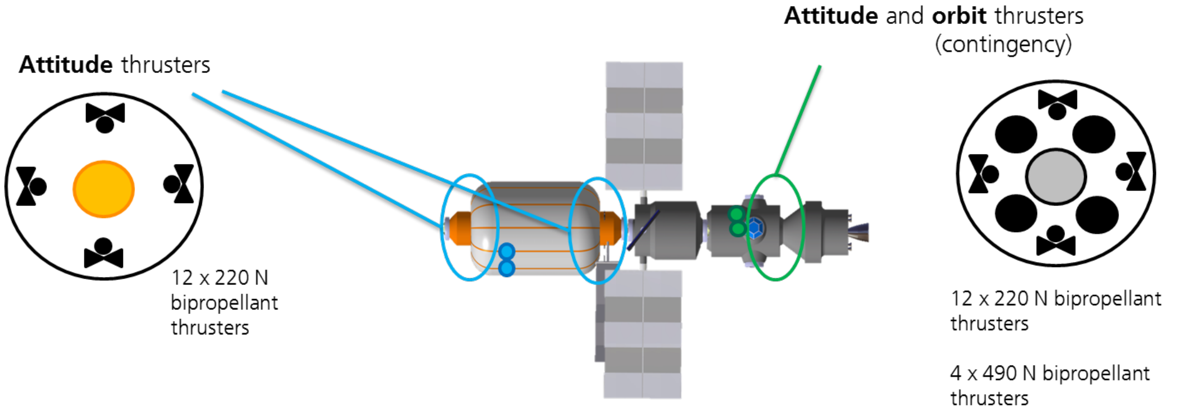


Figure 16-2: Baseline thruster configuration.

The designated function of each module for the above scenarios is provided in the table below.

Table 16-1: Activity overview of the different module's propulsion tasks with regard to the basic scenarios.

		Free Flyer	Habitat	SM/DN	Docked Vehicle
Scenario 1	*R&D/Attitude*				
	Debris Avoidance				
	Station Keeping				
	Torque counter				
Scenario 2	*R&D/Attitude*				
	Debris Avoidance				
	Station Keeping				
	Torque counter				
Scenario 3	***R&D/Attitude***				
	Debris Avoidance				
	Station Keeping				
	Torque counter				
Scenario 4	*R&D/Attitude*				
	Debris Avoidance				
	Station Keeping				
	Torque counter				

	Module actively providing propulsion, must be refuelled
	Contingency; required for non-nominal events but not refuelled
	Module not considered/present
	Propellant for assembly/initial docking only

The layout of the thrusters on the Base Station modules are therefore driven by the assembly scenario, however must also enable the contingency case. It was assumed that the propulsive requirements during Scenario Four would be met by the SM/DN; a discussion of alternative options is provided later in this chapter. The selected layout of the thrusters is shown in Figure 16-2 (in this case, assuming the SM and DN can be launched as one module).

The calculated propellant budget is outlined below.

Table 16-2: Propellant budget of the modules.

	Habitat	SM/DN	Visiting Vehicle (Electric Propulsion)	Visiting Vehicle (Chemical Propulsion)
Attitude (approach and/or docking)	600 (1)	550 (1)	0	3000 (1)
Contingency (torque countering)	70 (3)	70 (2)	0	0
Attitude (close-proximity)	70 (3)	*162 (1)*	0	*1200 (6)*
Station Keeping	0	0	109 (4)	0
Debris Avoidance	0	0	0	200 (5)
Total	**740**	**782**	**206**	**4400**
Total refuel (per 6 months)	0	232	206	4400

Assumptions:

- (1) ATV-based (ΔV)
- (2) Assumed moment arm of 25 m
- (3) Tank residuals; assumed sufficient for contingency
- (4) Value from mission domain
- (5) 6 manoeuvres/year; each ΔV = 1m/s

A bi-propellant (MMH/MON) propulsion system was selected considering possible restrictions to the use of toxic hydrazine propellant in the coming decade.

A schematic of the propulsion system for the SM/DN is shown below. Note the four clusters of three attitude thrusters, and the larger orbit-raising thrusters are all fed from the same propellant and oxidiser tanks.

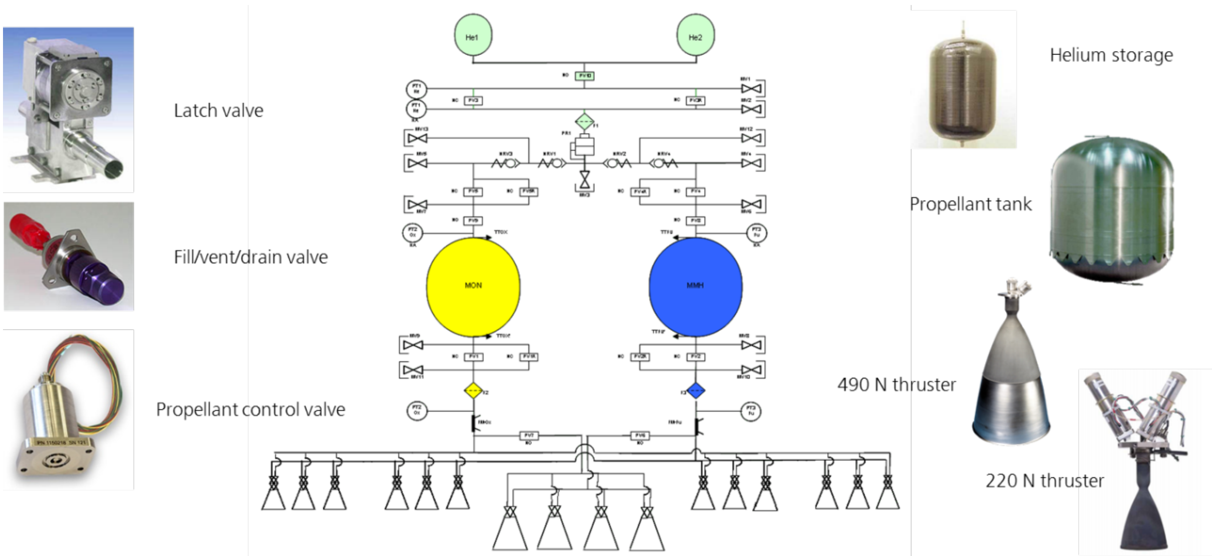


Figure 16-3: Design of the propulsion system of the Docking Node.

A schematic of the propulsion system for the Habitat is shown below.

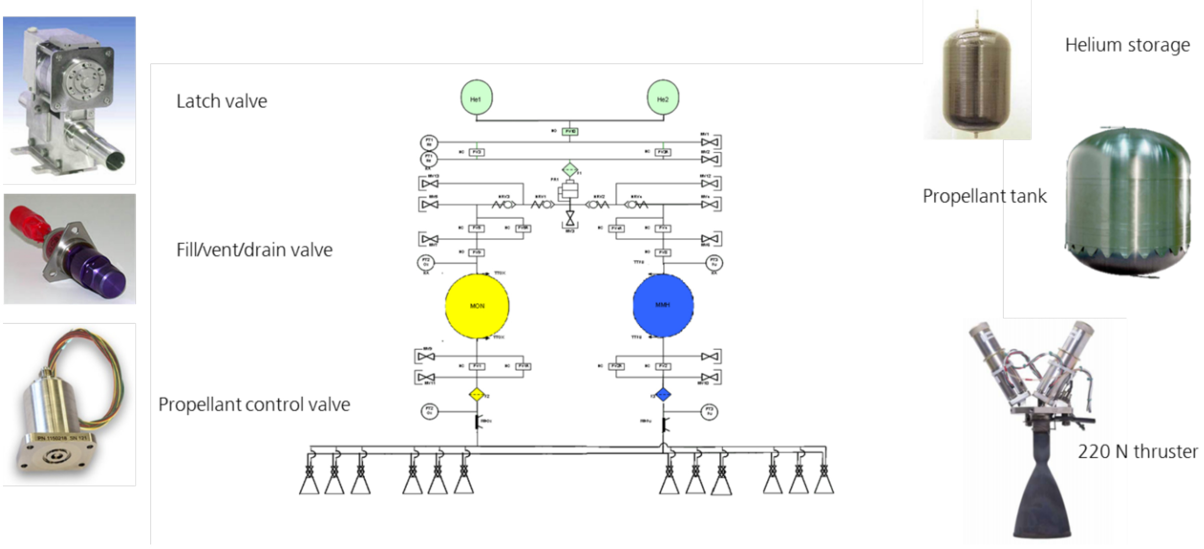


Figure 16-4: Design of the propulsion system of the Habitat module.

The technical specifications for the COTS thrusters employed in this system are shown below.

Table 16-3: Specification of the baseline thrusters [RD 18].

		200N (Astrium/ SNECMA)	R-4D-11 Aerojet
Thrust	*N*	216	490
Isp	*s*	270	312
Flow rate	*g/s*	78	141
Expansion Ratio	-	50	164
Fuel	-	MMH	MMH
Oxidiser	-	MON	MON
Mixture Ratio	-	1.65	1,65
Nozzle diameter	*mm*	95	279.4
Mass	*kg*	1.9	3.46

16.3. Options and Trades

16.3.1. Electrical vs. chemical propulsion

The propellant requirement for orbit-raising is very large and consequently the use of electric propulsion for orbit-raising has been investigated. The current ISS orbit-raising uses docked vehicles and the Zvezda service modules' two 3070 N N_2O_4/UDMH thrusters. The required propellant mass for orbit-raising and refuelling of the Zvezda is significant; e.g. ATV had 4700 kg propellant for re-boost and 860 kg propellant for refuelling.

Electric propulsion is an interesting alternative due to high Isp of the order 2000 – 4000 s (see figure below). The higher Isp required a smaller propellant mass, however much longer thrusting times.

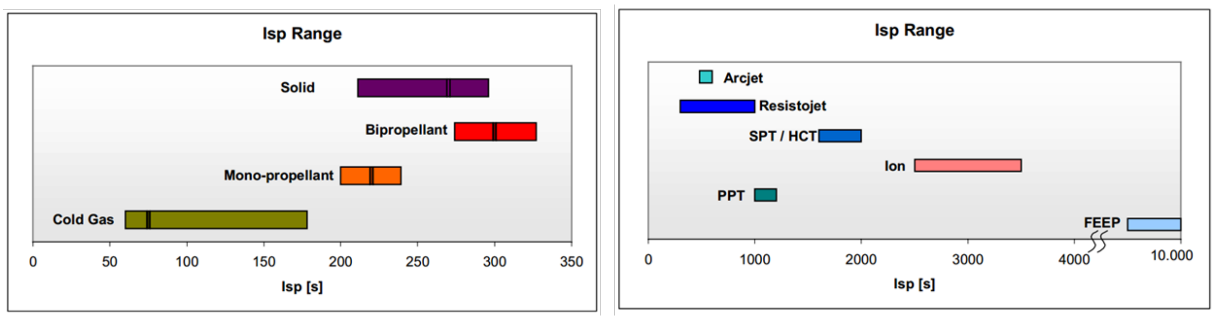


Figure 16-5: Comparison of specific impulse of different thruster types (left: chemical; right: electrical).

A qualitative evaluation of different propulsion systems feasibility to perform specific functions was performed, and is outlined in the table below.

Table 16-4: Qualitative evaluation of thruster types regarding functions.

	Mono	Bi-Propellant	Cold Gas	Electric
Rendezvous and Docking	Proven, modular, wide thrust range.	Proven, modular, wide thrust range.	Simple, safe, low Isp, low density	Novel solution; however high-thrust chemical system required for non-nominal events
Orbit Raising	Proven. Modular. Low Isp.	Proven. Modular. Low Isp.	Very low Isp.	High Isp; correct placement can allow for radiator-imposed drag-torque counterbalancing.
Debris Avoidance Manoeuvres	Proven, modular, wide thrust range.	Proven, modular, wide thrust range.	Simple, safe, low Isp, low density	Long manoeuvre time; unsuitable in case of late detection

Three options were examined further. The first of these options involved the use of high-TRL bi-propellants for all needs except for orbit-raising, which would be performed by a docked vehicle using electric propulsion. The latter two of these options consisted of multi-faceted systems, in which all propulsive requirements were met by the Base Station itself by one of two separate, on-board systems.

Table 16-5: Quantitative evaluation of propulsion system concepts.

	1	2	3
Description			
Rendezvous and Docking	Bi-propellant	Bi-propellant	Cold Gas
Orbit Raising	Docked vehicle	Electric Propulsion	Electric Propulsion
Debris Avoidance	Bi-propellant	Bi-propellant	Electric/Cold Gas
Mass (dry)	0	1	1
Mass (propellant)	-2	2	0
Power	2	0	0
Complexity	2	-2	-1
Risk	2	1	-2
Total	**4**	**2**	**-3**

It was found that despite the high mass savings it offered, the system complexity and power consumption requirements of the electric propulsion system made it prohibitive to be used as the baseline unless it was located on the docked vehicle. The main justification for this decision is the refuelling requirements. The high-pressure propellant (normally Xenon) tanks cannot be easily refuelled. Storage pressure for electric propulsion propellant (Xenon) is 200 – 300 bar. NASA is investigating ways to robotically refuel these tanks on-orbit, however TRL is very low. One option that was considered to mitigate this need is to use replaceable tanks. An extension of this would see the entire propulsion module replaced (limited lifetime thrusters makes this preferable). However, robotic or EVA operations would be required with this configuration, therefore it was discarded.

Electric propulsion on visiting vehicle was therefore selected as the best solution.

16.3.2. Orbit-keeping thrusters on Habitat vs. on Docking Node

The baseline configuration uses the propulsion system on the SM/DN to perform the orbit-keeping, attitude, debris avoidance and torque-countering manoeuvres in the event of the contingency scenario (Scenario Four). The other option, employing the

Habitat's propulsion system, was also considered. The decision to use the SM/DN system is herein justified.

The two options for the system are shown in the figure below.

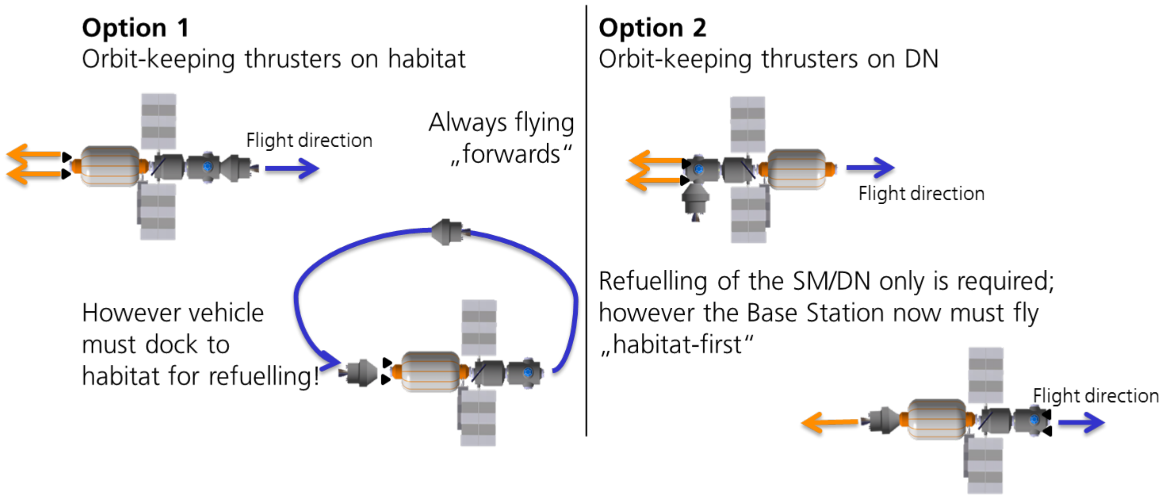


Figure 16-6: Options for the utilised contingency case propulsion system. Here, "orbit-keeping" refers to all contingency propulsion system needs.

Option 1 has the following advantages:

- The Base Station shall have the desired orientation during the flight phases not "Habitat-first".
- Supplied near Free Flyer; experimental equipment easy to deploy.
- One variation of this option is to keep the docked vehicle at the habitat at all times to increase simplicity and enable the desired flight direction. The Free-Flyer must then be docked to the DN side.

Option 1 has the following disadvantages:

- Habitat must be refuelled; therefore visiting vehicle must dock to habitat directly. This can only be performed during flight phases when the Free Flyer is not docked to the Base Station. This increases system and operations complexity.
- Restricted space for integration of propellant tanks and extra thrusters on the habitat.
- Supplies must be passed through habitat.

Options 2, involving the contingency needs being fulfilled by the docking node propulsion system, has the following advantages:

- Extra propellant tanks and propellant can be accommodated in the SM or DN (sufficient space should be available).
- More complex propulsion system on SM/DN increases the similarities and therefore production/qualification ease with the Free Flyer module.

Option 2 has the following disadvantages:

- Undesirable flight direction.

Considering these factors, the DN propulsion system shall be utilised in the contingency case. Note that all thrusters in the DN propulsion system are connected to the same tanks (see previous figures) so only two refuelling lines are needed (MON and MMH; with the option also for He). Furthermore, any thruster combination or cluster could be used, although it is foreseen that the large 490 N thruster shall be employed for all needs except fine-attitude control.

The extension of the system to include additional capabilities for the refuelling of the Habitat can be considered however is not selected for the baseline design. Doing this would allow a modular, flexible system that does not constrain the operations. The penalties would be with respect to mass (approximately 145 kg dry mass due to additional tankage and thrusters, Helium gas and refuelling fittings) and overall complexity.

16.4. Mass and Power Budget

The total dry mass for the propulsion system (both Habitat and SM/DN systems) is 742.9 kg, and 835 kg including margin. This includes all lines, fitting, electric equipment, valves and regulators, tanks, thrusters and pressurant gases.

Additional masses (not included in this budget) include 814 kg propellant mass in the Habitat (including 10% margin); and 938.4 kg propellant in SM/DN (including 20% margin).

16.4.1. List of Equipment

Table 16-6: Mass budget of the propulsion equipment Here, SM refers to the combined SM/DN module.

	Mass w/o margin [kg]	Margin [%]	Margin [kg]	Mass with margin [kg]	% of total dry mass
HA_oxidiser_tank	25.00	10.00	2.50	27.50	3.29
HA_pressurant	139.40	10.00	13.94	153.34	18.36
HA_propellant_lines_management	98.00	20.00	19.60	117.60	14.08
HA_propellant_tank	25.00	20.00	5.00	30.00	3.59
HA_Thrusters_220N	45.60	5.00	2.28	47.88	5.73
SM_oxidiser_tank	25.00	10.00	2.50	27.50	3.29
SM_pressurant	221.90	10.00	22.19	244.09	29.23
SM_propellant_lines_management	98.00	20.00	19.60	117.60	14.08
SM_propellant_storage	25.00	20.00	5.00	30.00	3.59
SM_propellant_tank	25.00	10.00	2.50	27.50	3.29
SM_Thrusters_220N	22.80	5.00	1.14	23.94	2.87
SM_Thrusters_490N	17.20	5.00	0.86	18.06	2.16
Total dry mass:	742.90			835.01	

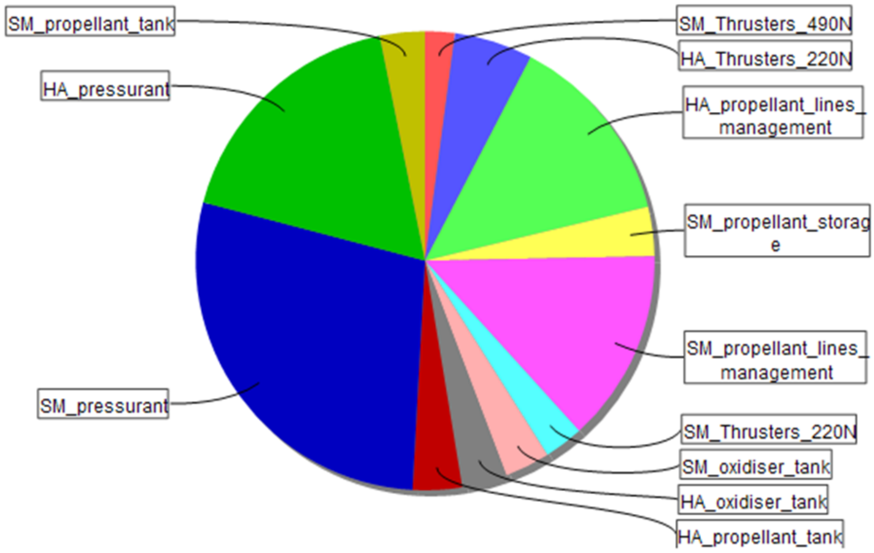


Figure 16-7: Pie chart of mass distribution of the propulsion equipment.

16.4.2. Power Budget

The power budget for the propulsion system is shown below. Note that this does not include the power requirement of any on board computers required for the operation of the propulsion system, and involves electric operation of valves, regulators and ignitors.

Table 16-7: Power budget of the propulsion equipment.

System Component	Parameter	Unit	Default	Standard_elect...	CrewExchange	Docked	Survival	Proximity_Ops
Propulsion	power_avg_wMargin	Watt	1364.780	1364.780	1364.780	1364.780	1364.780	1364.780
SM_Thrusters_490N	power_avg_wMargin	Watt	8.820	8.820	8.820	8.820	8.820	8.820
SM_Thrusters_220N	power_avg_wMargin	Watt	27.300	27.300	27.300	27.300	27.300	27.300
SM_propellant_lines_manager	power_avg_wMargin	Watt	38.400	38.400	38.400	38.400	38.400	38.400
HA_propellant_lines_manager	power_avg_wMargin	Watt	1248.000	1248.000	1248.000	1248.000	1248.000	1248.000
HA_Thrusters_220N	power_perUnit_avg_wMa...	Watt	27.300	27.300	27.300	27.300	27.300	27.300
SM_pressurant	power_perUnit_avg_wMa...	Watt	7.480	7.480	7.480	7.480	7.480	7.480
HA_pressurant	power_perUnit_avg_wMa...	Watt	7.480	7.480	7.480	7.480	7.480	7.480
Propulsion	power_energy_wMargin	Joule	1650837888.000	1650837888....	825418944.000	825418944.000	471667968.000	117916992.000

16.5. Re-Supply Items

The main item for the propulsion system is additional propellant for station keeping. This mass shall not be considered as payload of the visiting vehicle, rather should be accounted for in the vehicle propellant budget. There are three main classes of propellant for resupply (not including propellant required by visiting vehicle for rendezvous and docking etc.):

- Bipropellant for the refuelling of the SM/DN, to be provided as required. Estimation for this mass is 232 kg per 6 months.
- Propellant for high-thrust manoeuvres that cannot be performed by electric propulsion (debris avoidance, attitude control etc.),
- Propellant for orbit-raising (counter drag-induced altitude loss); can be performed by an electric propulsion system.

The baseline option is Xenon for electric propulsion station keeping. The requirement is 206 kg per 6 months. This was calculated assuming four (two operational, two contingency) RIT-22 thrusters (see below table for specifications). Xenon tank mass for this amount of propellant would be is 25 kg (1.4m ϕ).

RIT-22

Characteristic	RIT-22
Propellant:	Xenon
Ionisation principle:	Radio frequency excitation
Discharge chamber diameter:	22 cm
Beam voltage (nom):	900 to 2200 V
Beam current (nom):	750 to 2500 mA
Power (nom):	4.5 kW at 150 mN and 4400 sec. Isp
Thrust level:	75 to 175 mN (nominal) 40 to 250 mN (demonstrated)
Specific impulse:	3000 to 5500 sec. (nominal) 2500 to 6400 sec. (demonstrated)
Design Life	> 26,000 hours
Overall length:	23 cm
Outer Diameter:	30 cm
Mass:	7 kg

Figure 16-8: Specification of electrical thruster.

16.6. To Be Further Studied / Additional Considerations

There are several points that need refining, primarily the propellant budget. Activities to be performed are:

- Iteration of the propellant consumption with new station masses,
- Determine attitude ΔV requirements (all cases),
- Validation of estimations.

Additional trade studies should also be performed. These are, namely:

- Trade studies between propellant types (chemical AND electrical),
- Control scenario and assembly scenario,
- Use of Habitat propulsion system for contingency case.

16.7. Summary

The demands on the Base Station propulsion system were reduced by the decision to use the Free-Flyer and visiting vehicles for all nominal propulsion cases. The main requirements for the propulsion systems on the Habitat and the Free-Flyer are generated by the assembly phase, where (although passive targets in the docking) station-keeping capabilities are required. However, for the contingency case in which neither the Free-Flyer or a respective vehicle is docked and/or operational, propulsive manoeuvres shall be performed by the SM/DN propulsion system. To enable this, larger 490 N thrusters have been integrated on the SM/DN propulsion system in addition to the 12 200 N attitude thrusters. Furthermore, refuelling capabilities have been integrated into the SM/DN system.

17. Launch Scenario

17.1. Requirements and Design Drivers

The Base Station shall be assembled from two or three modules. These are:

- Habitat: 20 t
- Service Module (SM): 8.6 t
- Docking Node (DN): 10.7 t (combined with above SM in baseline solution).

It shall be assumed that crew exchange occurs every 6 months. The station's nominal orbit is specified to be:

- Orbit altitude of 400 km +/- 50 km,
- Orbit inclination of about 51.6 degrees.

17.2. Baseline Design

The baseline launch scenario is as follows:

- Launch 1: Free Flyer — Not within scope of CE study
- Launch 2: Habitat — Falcon 9 Heavy with Extended Fairing
- Launch 3: SM/DN — Atlas 551. Alternatives: Atlas 552, Ariane 6

The masses and dimensions of the modules, particularly the Habitat, are large but should not be prohibitive as future planned launchers should have the required capacity.

The option of launching the SM and DN separately is also considered (see following subsection).

17.3. Options and Trades

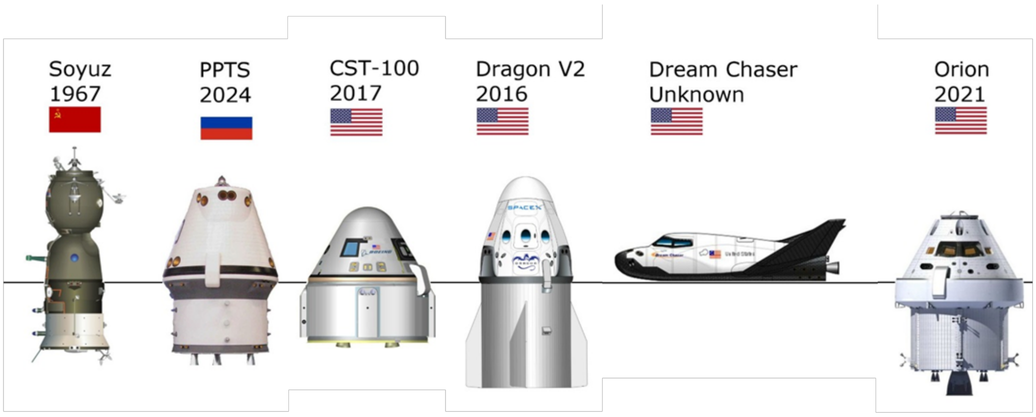
17.3.1. Launchers and Transportation Vehicles

An overview of viable launch vehicles is shown in the table below. The value for A64 was calculated based on available data.

Table 17-1: Overview of launcher capabilities [RD 19]

		Falcon 9	Falcon 9 Heavy	Atlas 551	A64
LEO altitude	*km*	200	200	185	*400*
LEO inclination	°	28,5	28,5	28,5	51,6
LEO payload	*kg*	13150	53000	18814	*18700*
Fairing diameter	*m*	5,2	5,2	5	*5,4*
Fairing height	*m*	13.9	13.9	26.5	*19.8*
Launch site	-	KSC, VAFB, CCAFS	KSC, VAFB, CCAFS	VAFB, CCAFS	*Korou*
Launch cost	*USD*	50-65	85-125	135-185	*65-90*

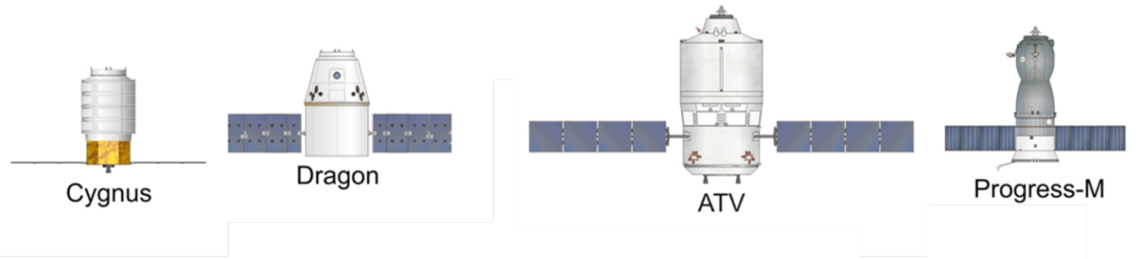
Options for crew transportation vehicles are shown below. The CST-100 and Dragon V2 are currently the expected baseline options.



		CST-100	Dragon V2	Dream Chaser	MPCV
Crew number	-	7	7	7	4
Launch vehicle	-	Atlas V	Falcon 9	Atlas V	Delta IV Heavy, SLS

Figure 17-1: Overview of crew transportation vehicles.

Options for supply vehicles are shown below. The Cygnus and Dragon V1 are currently the expected baseline options.



		Cygnus	Dragon V1	ATV	Progress
Payload Mass	*kg*	2700	6000	7670	2230
Launch vehicle	-	Atlas V	Falcon 9	Ariane 5	Delta IV Heavy, SLS

Figure 17-2: Overview of cargo vehicles.

17.3.2. SM and DN separately

The baseline option involves the launch of the SM and DN in one combined module. If the requirement on the synergies between the DN and the Free-Flyer node are relaxed, these can be combined into one module, resulting in significant mass savings. However, in the current design, they are still considered to be two separated modules that are launched together. If the masses and dimensions of the combined modules are prohibitive for simultaneous launch, they can be launched as separate modules. The mass of the systems would then increase slightly, with additional masses for the auxiliary power, AOCS and propulsion systems then being required.

The launch scenario would then be as follows:

- Habitat: 20 t
- Service Module (SM): 8.6 t + additional mass for power, AOCS, propulsion
- Docking Node (DN): 10.7 t.

With the following launch vehicle options:

- Launch 1: Free Flyer — Not within scope of CE study
- Launch 2: Habitat — Falcon 9 Heavy with Extended Fairing
- Launch 3: SM — Atlas 551. Alternatives: Atlas 552, Ariane 62
- Launch 4: DN — Atlas 551. Alternatives: Atlas 552, Ariane 62

17.4. To Be Further Studied / Additional Considerations

The following must be considered for the launcher scenario:

- Confirm availability of launch vehicles/crew vehicles,
- Confirm module masses and volumes, and
- Cost assessment must be performed; particularly regarding extra costs for e.g. new fairing development, and commercial use of crew vehicles.

17.5. Summary

The Base Station modules' dimensions and masses are not currently seen as prohibitive, with the launch scenario having no further impact on the design other than the two following aspects:

- The use of an extended fairing is a crucial enabler for the launch of the habitat. To be confirmed through ongoing negotiations with SpaceX,
- The mass of the combined SM/DN may exceed limits and necessitate additional launches.

The crew transportation system can also not be defined at this early stage, and development of commercial crew vehicles by Sierra Nevada, Boeing and SpaceX shall be observed closely.

18. Cost

18.1. Methodology

For space system cost estimations, there is four major common methods:

- Analogy (now) → Compare to and/or derive from existing, similar systems,
- Parametric (later) → Use statistical data relations, based on historical data,
- Bottom-up → Define and "built-up" cost per work packages and items,
- Expert Judgements → Consideration of expert's opinions; continuous task.

As can be seen in Figure 18-1, the different methods are more applicable in one project phase than in another. Due to the early stage of the project, no homogenous data is available. It means that some subsystems can already give rather detailed information (e.g. when they use standardized or heritage items) whereas others have to be developed from scratch. This would make bottom-up estimations difficult. Since the very few existing 'human spaceflight' cost estimation relationships (CERs) still have to be adjusted, parametric cost estimation will be used throughout the later project phases as well. For now, the analogy-based assessment has been the preferred method, using mainly ISS-related information to scale and compare cost between the existing station and the here proposed successor.

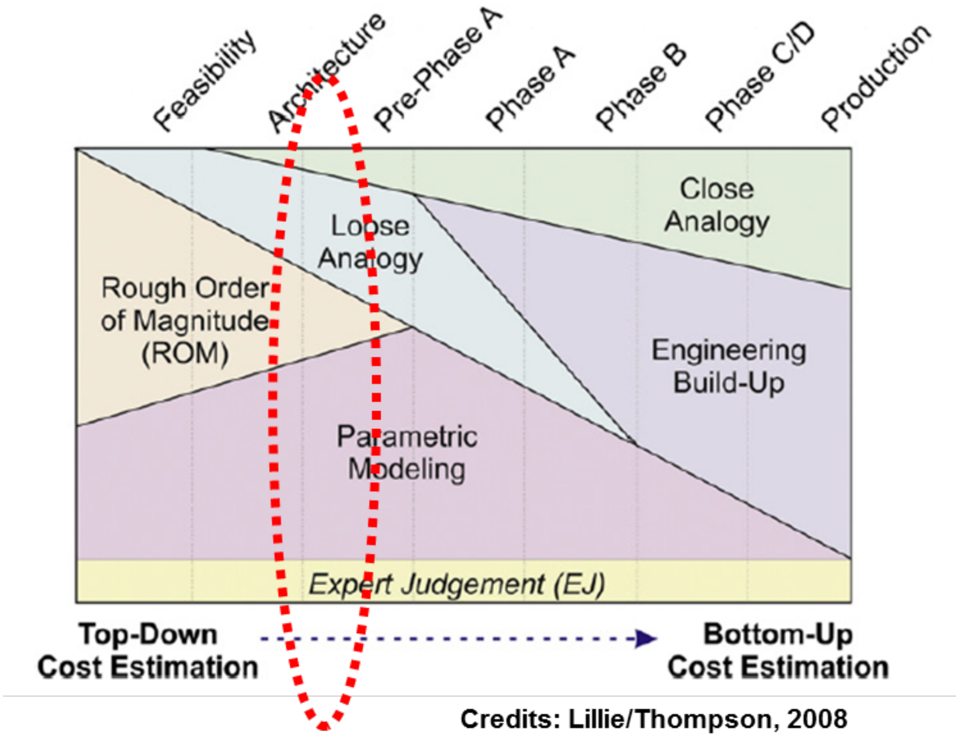


Figure 18-1: Cost estimation methods related to the project phase.

18.2. Requirements and Design Drivers

For this cost estimation, following assumptions (rather than requirements) have been used with respect to both, development and operational expenses:

- There is always a 3 people crew on the base station, which will be exchanged every 6 months (based on decision made during the study on 10.06.15)
- The built-up phase (here called "year 0") includes:
  - Modules cost
  - Estimated launch cost of cargo and crew, incl. 1 year of operations,
  - 1 crew, with 6 month stay only (i.e. in the last half-year of "Year 0").
- Year 1-n cost for station operations ("Ops"):
  - Continuous Mission operation
  - 2x crew & cargo transport each
- The cost in this report are mainly stated in Billion USD, where the unit „Billion" is the equivalent to the German „Milliarden" (plus considering exchange rate).

18.3. Reference Station

As already stated in chapter 18.1., the main reference for the present analogy-based cost assessment is the International Space Station (ISS). Independently of the size, it has more than 40 modules which is about 10 times more than the Post-ISS base station. Figure 18-2 provides a simplified comparison of the various modules and furthermore indicates which are not yet relevant for the current estimates (see orange modules).

ISS

- 5 „Labs", Lab mods. (C,D,N,L;JEM)
- 3 research modules (MRMs, MLM)
- 6/8 big Truss Segments
- 3 small Truss Sements
- 3 ‚other' service modules (Z, JEM, SM)
- 4 exPress Logistic ctr. (small)
- 3 ext. storage plats (smaller)
- 3 nodes (U,H,T)
- 3 arm modules (ESA,Canada,JEM)
- 4 adapter & airlocks (3xPMA,Quest)
- 1 Cupola
- 1 Alpha Mag. Spectrometer
- 1 SPDM Dextre
- 1 MBSMT
- 1 Funtional cargo block
- (8 S/A on big truss segments, Star & Port)

Post-ISS baseline

- Expandable Habitat
- Service Module
- Docking Node
- EVA Module
- (Visiting vehicle)

→ 4 modules vs.
→ ~44 modules (ISS)

Figure 18-2: Comparison of ISS and Post-ISS Scenario-I modules.

18.4. Adjusted Analogy-based Cost Estimate

For the comparison, the mass per module and cost per mass of the ISS have been identified:

- ISS mass/module factor: 10 t (in average)
- ISS cost/mass factor: 0.08 billion $ per ton

The ISS module development cost have been derived primarily from the "GAO Space Station U.S. Life-cycle Funding Requirements Testimony" [RD 20], and are assessed with around 35 billion $. An initial assumption taking into account the mass/module-factor led to a rough order of magnitude (ROM) cost for the base station of ~3.8 billion $.

In the course of the present study, the Post-ISS base station mass turned out to be ~63 t instead of the initially derived 40…48 t. As can be seen in Table 18-1, this leads to a total modules cost of ~5.1 billion $, including development and first unit cost.

Table 18-1: Analogy-based development cost estimation ISS vs. Post-ISS Scenario-I.

	ISS	Post-ISS
No of modules	~44	4
Mass [t]	450 t (10 in average)	63 t * (26 Habitat) (22 Service) (17 Node) (0 EVA)
Module cost [billion $]	~35	~5,1 **

** Based on Post ISS mass*
*** Based on derived ISS cost/mass factor: 0,08 [billion $/ ton]*
*Reduced **to 3,5 billion** with B330 price of 0.36 billion $ instead of 2 (w/ 0.08 ratio)*

However, since the project plans to make use of a habitat module provided by Bigelow Aerospace, located in Las Vegas, USA, the current estimate does not include the habitat development but only the first unit cost. This will be part of future negotiations.

For now it can be assumed that the habitat first unit cost is around the usual 15…20% of the overall development cost. This would reduce the total amount to **~3.5 billion $.**

18.5. Annual Cost

After "Year 0", when the Post-ISS starts its regular operations, the recurring cost will dominate the annual budgets. So far, the system operational cost ("SysOps") and the launches for crew and cargo replacements are taken into account; which are assumed as:

- 0.1 billion $ for the station mission operations
- 0.05 billion $ for additional habitat operations
- 2 crew launches per year
- 2 cargo launches per year

Comparing the numbers (as done in Table 18-2) with a NASA audit report on extending the Operational ISS Lifetime until 2024 [RD 21] it can be seen that there are additional cost, which are not defined yet for Post-ISS. This will further increase the current estimate.

These increases include future station replacement-items (e.g. new laptops) or additionally required ground-based expenses. In summary, both estimates (the official ISS-one as well as the presented study results) include rather optimistic values.

Table 18-2: Analogy-based estimation of operational cost ISS vs. Post-ISS Scenario-I.

ISS Ref for OPS TOTAL [Bio $]*		
Total	**2,9**	**100%**
SysOps	1,23	43%
TranspCost	0,97	34%
Crew cost	0,37	13%
Cargo cost	0,60	21%
Research	0,273	10%
Labor/Trav	0,223	7%
Cargo/Crew	0,167	6%

Post-ISS for OPS TOTAL [Bio $]*		
Total	**0,85**	**100%**
SysOps	0,15	18%
TranspCost	0,70	82%
Crew cost	0,45	53%
Cargo cost	0,25	29%
Research		0%
Labor/Trav		0%
Cargo/Crew		0%

***SysOps:** *Mission Ops, H/W incl Orbital replacement units (orbit and ground; see ISS laptops), EVAs*
***TranspCost**: all transport expenses; currently buying seats in other capsules & cargo re-supply contracts*
***Research:** research cost aboard*
***Labor/Trav:** labor, salary, travel and personnel expenses for civil servants to ISS program,*
***Cargo/Crew:** station development projects such as Int.- Docking Adapter or Common Comms Box*
→ source: NASA Audit report IG-14-031 for extending Ops Life of ISS until 2024; September 18, 2014

18.6. Cost over Time

In order to calculate the total program cost for e.g. 10 years, the development, built-up and operational cost need to be consolidated.

The resulting bar chart presented in Figure 18-3 is based on following elements:

- Year 0 development cost = 3.1 billion $ (Docking Node, Service Module)
- Year 0 procurement cost = 0.4 billion $ (Habitat price; TBC)
- Year 0 launch cost = 0.7 billion $ (2x cargo, 1x crew)
- Year 0 "SysOps" cost = 0.15 billion $
- Annual Operation & Launch cost = 0.85 billion $ starting in Year 1
- Inflation of 3% per year

Independently of the launch scenario, the cost-related estimate for the build-up phase includes two Falcon-9 heavy launches with a modified, extended fairing which is a potential option of the future joint Space Exploration Technology Corporation & Bigelow Aerospace service portfolio.

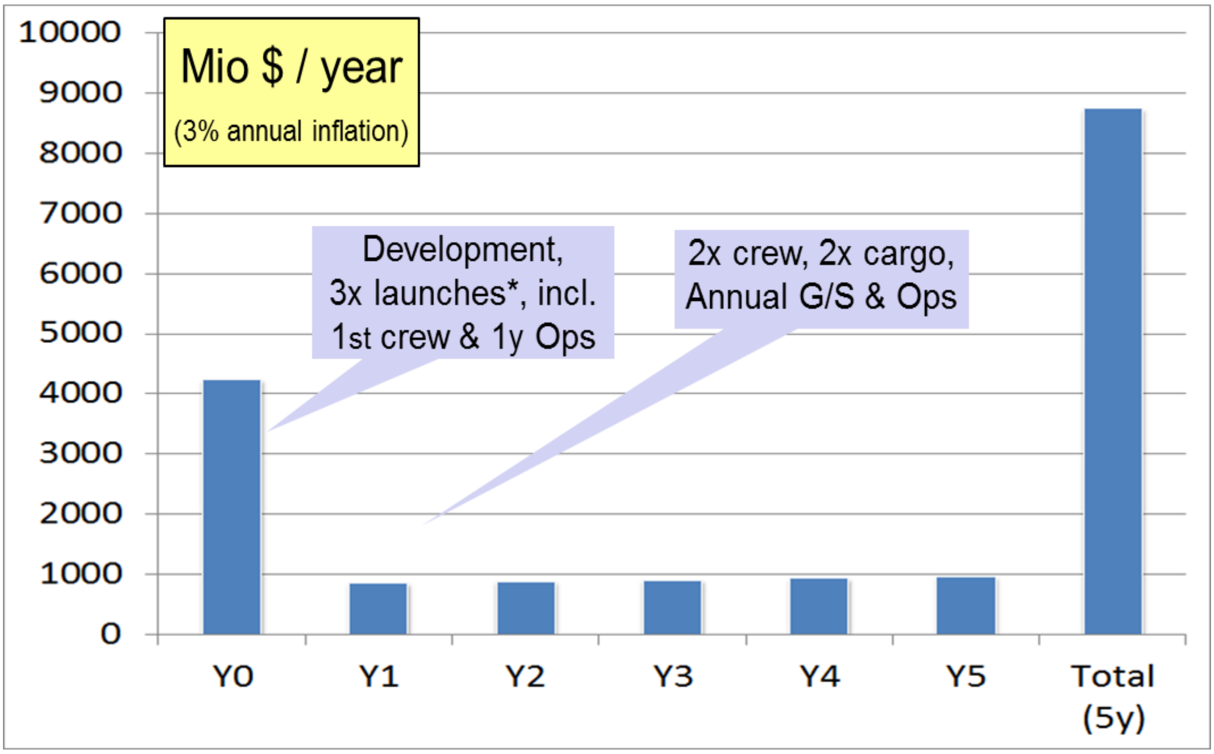


Figure 18-3: Cost over time for Base Station (mainly analogy based).

18.7. Summary

The present "Post-ISS base station – Scenario I" cost estimated includes, as already shown in chapter 18.6, around 3.5 billion $ for development and procurement and around 1 billion $ per year (in average, starting in "Year 0") for operational cost, taking annual inflation into account.

This concludes, that:

- after 5 years the cumulative cost are ~9 billion $,
- and after 10 years cost sum up to: ~16 billion $,
    - which is the equivalent to: ~14 "Milliarden" €
      (based on the currency exchange rate on June 12th, 2015)

It is important to note that

- All values are based on preliminary adjusted analogy estimate,
- Cost numbers from Bigelow are heavily "to be clarified"
- Cost are likely to increase, since many small items are not covered yet.

18.8. To Be Further Studied

In general, there are – due to the early phase of this study – many unknowns which require further analyses. For the on-going cost estimation, the most obvious and impacting aspects are:

- The Launch Vehicle (LV) selection (and launch scenario),
    - incl. the cost per LV provider,
        - which could be either assessed as „seat-based"
          (as it is the case for current Soyuz launches), or
        - considering own capsule development- respectively procurement cost & the full transportation responsibility (i.e. the launch itself).
- Potential replacements to the station, which defines cargo transport per year.
- Additional cost due to ground-based work (as mentioned in chapter 18.5).
- Detailed development cost by more elaborated parametric analyses (in Phase A).
- and subsequently the continuous and iterative update of the assumptions made related to cost/mass-factors, launch and operations.

19. Acronyms

Abbreviation	Comments
ATV	Automated Transfer Vehicle
BCDU	Battery Charge Discharge Unit
BS	Base Station
CE	Concurrent Engineering
CEF	Concurrent Engineering Facility
CER	Cost Estimation Relationships
CIGS	Copper-Indium-Gallium-Selenide
CMG	Control Momentum Gyro
COTS	Commercial off-the-shelf
CSA	Canadian Space Agency
DC	Direct Current
DCSU	Direct Current Switching Unit
DDCU	DC/DC Converter Unit
DLR	Deutsches Zentrum für Luft- und Raumfahrt e.V.
DN	Docking Module
DOD	Depth of Discharge
DoF	Degrees of Freedom
ECLSS	Environmental Control and Life Support System
EDRS	European Data Relay System
EOL	End of Life
ESA	European Space Agency
EVA	Extra Vehicular Activity (Space Walk)
FF	Free-Flyer
FOV	Field of View
GNSS	Global Navigation Satellite System
HA	Habitat Module
HST	Hubble Space Telescope
IBDM	International Berthing and Docking Adapter
IMU	Inertial Measurement Unit

ISPR	International Standard Payload Rack
ISS	International Space Station
ITAR	International Traffic in Arms Regulations
KSC	Kennedy Space Centre
LEO	Low Earth Orbit
MBSU	Main Bus Switching Unit
MDM	Multiplexer Demultiplexer
MMH	Monomethylhydrazine
MON	Mixed Oxides of Nitrogen
MPCV	Multi-Purpose Crew Vehicle
MPPT	Maximum Power Point Tracker
NASA	National Aeronautics and Space Administration
NDS	NASA Docking System
OBC	On Board Computer
OPALS	Optical PAyload for Lasercomm Science
P/L	Payload
REoCV	Reduced End-of-Charge Voltage
ROM	Rough Order of Magnitude
RVDS	Rendezvous and Docking System
S/S	Subsystem
SLS	Space Launch System
SM	Service Module
SoC	State of Charge
STK	AGI Systems Tool Kit
TC	Transfer Converter
TCS	Thermal Control System
TDRS	Tracking and Data Relay Satellite
UDMH	Unsymmetrical Dimethyl Hydrazine

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